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PERFORMANCE ESTIMATES FOR POWERED PARAFOIL SYSTEMS

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13. ABSTRACT

The estimated flight performance of powered Parafoil systems are presented for various system weights ranging from 10 pounds to 10,000 pounds. Special attention is given to level flight, climbing and descending flight, and take-off distance. The equations of motion are summarized and the effect of the thrust pitch angle is evaluated. Various system designs such as manned flight, cargo and weapon delivery, and decoy and jammer vehicles are considered as examples for using the performance curves and equations.

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FOREWORD

This is the second report prepared by the University of Notre Dame, Notre Dame, Indiana under U.S. Air Force Contract F33615-71-C-1093. The first report was numbered AFFDL-TR-72-23 entitled Parafoil Powered Flight Performance. This contract was initiated under Project 6065, Performance and Design of Deployable Aerodynamic Decelerators, Task 6065 01, Terminal Descent Parachutes for Tactical Air Drop and Military Vehicle Recovery. The work was administered under the direction of the Recovery and Crew Station Branch (AFFDL/FER) of the Air Force Flight Dynamics Laboratory at Wright-Patterson Air Force Base, Ohio. Mr. R. Speelman served as project engineer during the duration of the effort.

The author, of the University of Notre Dame Aerospace and Mechanical Engineering Department, was Dr. John D. Nicolaides, Professor. The author is indebted to Joe McCarthy, for his assistance in the calculation and preparation of the performance curves.

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Publication of this report does not constitute Air Force approval of the report's findings or conclusions. It is published only for the exchange and stimulation of ideas.

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LIST OF SYMBOLS

α	angle of attack (deg)
γ	angle that the flight path makes with the horizontal (deg)
η	dimensionless thrust factor
$\underline{\eta}$	thrust ratio factor
θ	thrust angle; angle that the thrust line makes with the horizontal (deg)
μ	ground resistance factor
ρ	density of air (slugs/ft^3)
A	planform area of Parafoil airfoil (ft^2)
AR	aspect ratio
C_D	coefficient of drag
ΔC_D	additional drag coefficient due to size of vehicle and vehicle suspension system
C_L	coefficient of lift
D	drag force (lbs)
ΔD	additional drag due to size of vehicle (lbs)and vehicle suspension system
deg	degrees
fpm	feet per minute
fps	feet per second
ft	foot or feet
g	gravity
HP	horsepower
ΔHP	excess horsepower for calculation of rate of climb

L	lift force (lbs)
lbs	pounds force
L/D	aerodynamic lift to drag ratio
m	mass
mph	miles per hour
ND 2.0(400)	indicates a Parafoil with an aspect ratio of 2.0 and a planform area of 400 ft^2
q	dynamic pressure, $\frac{1}{2} \rho V^2$
R/C	rate of climb available under conditions being analyzed (fpm)
T	thrust (lbs)
T_o	static thrust ($T_o = T_\theta$ at $\theta = 0$)
u	horizontal velocity; velocity in x direction (fps)
V	total velocity (fps, mph) also U when horizontal
w	vertical velocity; velocity in z direction (fps)
W	weight (= mg) (lbs)
x	horizontal inertial axis
$\ddot{x}$	acceleration along x axis
z	vertical inertial axis
$\ddot{z}$	acceleration along z axis
X	take-off distance
X_T	true take-off distance
R	ground resistance
V_S	stall velocity

INTRODUCTION

The secret of flight is the wing and all of the wings of aviation have been rigid. The early fabric wings contained rigid members and, like the metal wings of today, they were rigidly attached to the fuselage and moved rigidly with it. The Parafoil, however, is a completely non-rigid wing. It is made entirely of fabric with absolutely no rigid members. It may be stuffed into a bag or folded into a pack. Since it is made entirely of plastic coated nylon, it is very light. A 400 square foot Parafoil wing, for example, weighs less than 20 pounds.

Not only is the Parafoil wing itself completely nonrigid but it is non-rigidly attached to the fuselage and, thus, the Parafoil wing and the fuselage may move independently.* In flight, this movement requires special consideration in the equations of motion used in computing the flight performance and stability. On the ground, this free movement and the non-rigidity provide a more convenient and flexible system for handling, packing, and storing.

It is clear, therefore, that the Parafoil represents an entirely new aviation concept which may have many interesting applications. Already the Parafoil has been used as a kite[5], as a sport jumping glider[5,10], as an ascending manned glider[7], as an air dropped guided cargo or weapons delivery system[3,11], as a gliding decoy system[14], as an aerial target system[3], as a recovery and gliding projectile system[14]...

The Irish Flyer[12] is a unique aircraft which uses the Parafoil and is flown like an airplane with an engine and propeller. A summary and background on the Irish Flyer is given in the technical report, "Parafoil Powered Flight Performance"[12]. It is the purpose of this report to set down in summary form the equations of motion for the flight performance of the Irish Flyer and to provide general performance curves for level flight, for climbing flight, and for take-off distance for various flight system weights and powers.

Specifically, the reader may easily determine for his proposed system preliminary estimates for the level flight velocity, the rate of climb, the take-off distance, the required horsepower,... etc. all depending upon his selection of Parafoil design (C_L, C_D, L/D, α_T ...), system weight, wing

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*When not in flight the Parafoil lies limply on the ground; however, when in motion the ram air entering the leading edge of the Parafoil provides inflation and rigidity. The lift from the Parafoil in flight produces tension in the shroud lines and thus the payload is carried through the air with flight system rigidity.

loading, etc,... The design curves include system weights ranging from 10 pounds to 10,000 pounds.

The original reason for undertaking the design of the Irish Flyer* was simply because it seemed like an interesting entirely new thing to do. The early powered Parafoil ** flights demonstrated that slowly descending flight was possible. Excellent flight stability and control was accomplished; however, insufficient horsepower prohibited climb. As the flight test program proceeded, it became clear that the Irish Flyer offered many interesting applications. The obvious immediate application is as an extremely cheap and safe sport flying vehicle. However due to the Vietnamese War and the heavy loss of pilots, attention was given to using the Irish Flyer as a flying ejection seat with a range of 100 miles. As a further application, there is a requirement to hunt down, jam, and destroy enemy radar and missile sites. Accordingly, consideration was given to using the Irish Flyer as an unmanned powered homing flight vehicle,[14] which can be carried externally on a flighter or helicopter and dropped like a bomb, (Special Modular Bomb). Another application is as a special reconnaissance vehicle which may be air dropped and then Remotely Piloted (RPV).[14] Other applications in air and also underwater have been proposed.

Thus, the demonstrated performance of the Irish Flyer *** from various flight tests and the increasing interest in various application areas all suggested that this report be prepared which would provide the interested designer with a rational basis for preliminary system design decisions and a basis for estimating powered Parafoil flight performance.

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*All rights to Powered Parafoil Applications and to the Irish Flyer concept are held by Dr. John D. Nicolaides, No. 437969, Patent Office.

**The Parafoil is a design and development of Dr. John D. Nicolaides (patent pending 105836), and is based on the multi-cell ram airfoil Patent No. 3285546.

***Dr. Nicolaides acting completely on his own authority undertook the personal design and construction of the flight test vehicles and personally carried out the associated flight test program under FAA/SAC Numbers N-3029 and N-302ND.

AERODYNAMIC DATA

The aerodynamic data for the Parafoil was obtained from two sources, wind tunnel tests and full scale flight tests.[1-4,7]

The wind tunnel tests were carried out at the University of Notre Dame, at the U. S. Air Force Flight Dynamics Laboratory, and at the National Aeronautics and Space Administration (Langley Field). The technical report, "Parafoil Wind Tunnel Tests", contains the wind tunnel test results.[4]

The full scale flight tests were carried out at the University of Notre Dame, at the U. S. Air Force Flight Dynamics Laboratory (Wright Field), and elsewhere. The technical report, "Parafoil Flight Performance" contains the principle flight test results.[7]

The aerodynamic data for the lift coefficient (C_L), the drag coefficient (C_D), and the lift-to-drag ratio (L/D), as used in this report, are given in Figures 1-5. Additional incremental drag coefficients may be used to account for any additional payload drag produced by the specific system configuration of special interest to the designer.*

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*The aerodynamic data used in this report was obtained from Ref. 7. The basic drag includes canopy, minimum line rigging and minimum payload. The line drag estimate is based on an area of 5.5 ft^2 and a drag coefficient of .6 (Fig. 18, Hoerner).[13] The payload drag estimate is based on an area of 2.5 ft^2 and a drag coefficient of .8.

For preliminary design estimates it is suggested that the total payload drag be estimated and added to the basic curve of this report. To assist the designer, two additional drags of .038 and .076 have been included.

An illustration of the Irish Flyer in general flight is given in Figure 6. In this general case the Irish Flyer may be climbing or descending ($\gamma \neq 0$) and the thrust line may not be coincident with the velocity vector or the horizontal ($\theta \neq 0$). The equations of motion are derived as,

$$T\cos\theta + L\sin\gamma - D\cos\gamma = m\ddot{x} \tag{1}$$

$$-T\sin\theta - L\cos\gamma - D\sin\gamma + W = m\ddot{z} \tag{2}$$

For steady state flight these equations reduce to

$$T\cos\theta + C_L q\, A\sin\gamma - C_D q\, A\cos\gamma = 0 \tag{3}$$

$$-T\sin\theta - C_L q\, A\cos\gamma - C_D q\, A\sin\gamma + W = 0 \tag{4}$$

The flight velocity of the Irish Flyer is obtained from Equation (3)

$$V = \left[\frac{T\cos\theta}{(C_D\cos\gamma - C_L\sin\gamma)\,{}^1\!/_2\,\rho A}\right]^{1/2} \tag{5}$$

and from Equation (4) as

$$V = \left[\frac{W - T\sin\theta}{(C_L\cos\gamma + C_D\sin\gamma)\,{}^1\!/_2\,\rho A}\right]^{1/2} \tag{6}$$

or

$$V = \left[\frac{W - \frac{\eta W}{L/D}\sin\theta}{{}^1\!/_2\,\rho A\,(C_L\cos\gamma + C_D\sin\gamma)}\right]^{1/2} \tag{7}$$

where $\eta \equiv {}^T\!/_W \big/ {}^L\!/_D$ (7a)

For level flight, ($\gamma = 0$)

$$V = \left[\frac{2W}{\rho A}\left(\frac{1}{C_L + C_D\tan\theta}\right)\right]^{1/2} \tag{8}$$

For level flight ($\gamma=0$) and $\theta = 0$,

$$V = \left[\frac{2W}{\rho A C_L}\right]^{1/2} \tag{9}$$

The angle of climb or descent is obtained by equating Equation (5) and (6),

$$\gamma = \tan^{-1}\left[\frac{\left(1-\frac{T\sin\theta}{W}\right)-\frac{L}{D}\left(\frac{T\cos\theta}{W}\right)}{L/D\left(1-\frac{T\sin\theta}{W}\right)+\left(\frac{T\cos\theta}{W}\right)}\right] \tag{10}$$

or

$$\gamma = \tan^{-1}\left[\frac{1-\frac{\eta\sin\theta}{L/D}-\eta\cos\theta}{L/D\ 1-\frac{\eta\sin\theta}{L/D}+\frac{\eta\cos\theta}{L/D}}\right] \tag{11}$$

The horizontal and L/D vertical velocities of the Irish Flyer are given by

$$u = V\cos\gamma \tag{12}$$

$$w = V\sin\gamma = -\frac{R/C}{60} \tag{13}$$

Thus, the flight path angle, γ, of the Irish Flyer may be obtained from Equation (11) by inputting the numerical value of the thrust angle (θ), the lift-to-drag ratio (L/D) for a fixed flight trim angle of attack (α), and the thrust factor* η. The total velocity of the Irish Flyer may then be obtained from Equation (7) by inputting γ as obtained from Equation (11) and $C_L(\alpha)$ and $C_D(\alpha)$. The horsepower required for steady state level flight may be obtained from

$$HP = \frac{DV}{550} = \frac{(T\cos\theta)\,V}{550} \tag{14}$$

*The thrust factor is given by Equation 7a and may be evaluated by inputting the known thrust.

Thus, we are able to obtain the flight performance of the Irish Flyer from solutions of the large angle equations of motion.*

*The basic flight equations (Equation 3 and Equation 4) allow large angles of climb, large angles of attack, and large thrust angle. The development based on these equations (Equations 10 through 14) also allow flight performance evaluation at large angles. While the preliminary design curves of this report emphasize small angle of flight, it is important to note that the equations given therein may also be used for large angle flight.

PERFORMANCE CALCULATIONS

Level Flight Velocity

The Irish Flyer velocity in horizontal flight may be determined if the wing loading (W/A) and the lift coefficient (C_L) are known. The lift coefficient depends on the Parafoil aspect ratio and the flight angle of attack which is generally near the value for best L/D ($\alpha = 8^o \pm 2^o$).

A wing loading of one has been used quite extensively by military jumpers for some years. Also, a wing loading of one has been used in many Parafoil guided delivery systems. However over the years wing loadings ranging from .25 to 5 have been used with complete success and higher values are certainly possible.

Therefore in calculating the Irish Flyer level flight velocity at $\theta=0$ wing loadings ranging from 0 to 10 have been used and lift coefficients of .25, .5, .75, and 1.0 have been used.

The level flight velocity for wing loadings up to three are given in Figure 7. The increased level flight velocities achievable by increased wing loadings (0-10) are given in Figure 8.

Unlike the conventional airplane which is only able to fly over a limited range of angles of attack and can experience catastrophic stall, the Irish Flyer can fly over a range of angles of attack from -10^o to 90^o and does not experience stall.* It is of interest, therefore, to compute Irish Flyer level flight velocity over a larger range of angles of attack. Figure 9 represents these calculations for angles of attack from 0^o to 30^o for wing loadings of 1, 5 and 10.

Horsepower Required for Level Flight

The horsepower required for a given level flight velocity at $\theta=0$ may be calculated, if the Irish Flyer weight and system Lift-to-Drag ratio are known.

*A normal rigid wing at an angle of attack of near 20^o experiences sudden loss of lift called "stall". One of the primary advantages of the Parafoil is that it does not experience sudden loss of lift or stall; but rather its lift drops off slowly at rather large angles of attack.

$$HP = \frac{WV}{L/D\ 550} \qquad (15)^*$$

The horsepower required for a level flight velocity of 41 ft/sec ($W/A = 1.5$; $C_L = .75$) for Irish Flyers weighing 10 pounds to 50 pounds is given in Figure 10 as a function of lift-to-drag ratio.

Similar calculations for Irish Flyers weights from 50 pounds to 1000 pounds and from 2000 pounds to 10,000 pounds are given in Figures 11 and 12.

For an Irish Flyer level flight velocity of 58 ft/sec ($W/A = 3.0$; $C_L = .75$) and for weights from 10 pounds to 10,000 pounds, the horsepower required is given in Figures 13, 14, and 15, as a function of lift-to-drag ratio.

For a level flight velocity of 82 ft/sec similar results are given in Figures 16, 17, and 18.

Climbing Flight

The additional horsepower required for various rates of climb for various vehicle weights is given by

$$\Delta HP = \frac{W\ R/C}{550 \times 60} \qquad (16)$$

The additional horsepower required for Irish Flyer weights from 10 pounds to 10,000 pounds for rates of climb of 100, 200, 400, 600, 800, and 1,000 ft/min. is given in Figures 19, 20, and 21.

The rate of climb and the level flight velocity determine the angle of climb (γ). The FAA requires a R/C of 400 ft/min. If a light plane velocity of 60 MPH is assumed, then the angle of climb is 4.33°. Accordingly, the rate of climb and flight velocity for angles of climb of 2°, 4°, 5°, 6°, 8°, and 10° are given in Figure 22. Assuming that a 4° angle of climb is required, then Figure 23 provides the additional horsepower required for various vehicle weights and flight velocities.

*This equation is accurate for near level flight. For example at angles of climb or descent of 10°, an error of 1.5 percent is involved. At angles of 20° an error of 6 percent is involved.

Effect of Thrust Angle

For the previous Irish Flyer performance calculations the thrust line was taken as horizontal, ($\theta = 0$). However, it is possible to use other angles, both fixed and changeable in flight. Accordingly, the effect of θ angle is given in Figure 24a.

In the case of level flight the effect of thrust angle, θ, may be seen from,

$$\frac{T_o}{T_\theta} = \cos\theta + \frac{\sin\theta}{L/D} \tag{17}$$

which is obtained by solving Equation (4) for lift and substituted for lift as obtained from Equation (3)*, all for the case of $\gamma = 0$. Figure 24b illustrates the results from Equation (17) where it can be seen that significant reductions in thrust are obtained by introducing θ angle when the lift-to-drag ratio is low. For improved lift-to-drag ratios the advantage of thrust angle is seen to be markedly reduced.

The introduction of thrust angle also results in a reduction in flight velocity as given by,

$$\frac{V_o}{V_\theta} = \left[1 + \frac{\tan\theta}{L/D}\right]^{1/2} \tag{18}$$

and a reduction in horsepower as given by,

$$\frac{HP_o}{HP_\theta} = \left[1 + \frac{\tan\theta}{L/D}\right]^{3/2} \tag{19}$$

provided of course that the θ angles are in the range of the thrust ratios greater than one.

For example at $\theta = 25^o$ and $L/D = 2.5$, we obtain

$T_\theta = .929\ T_o$ or 7.1 % reduction in thrust

$V_\theta = .918\ V_o$ or 8.2% reduction in velocity

$HP_\theta = .773\ HP_o$ or 22.7% reduction in horsepower

---

*Drag = $L/_{L/D}$

However, at $\theta = 10^{o}$ and L/D = 5, we obtain only

$$T_{\theta} = .980 \ T_{o} \qquad \text{or} \qquad 2.0\% \text{ reduction in thrust}$$

$$V_{\theta} = .982 \ V_{o} \qquad \text{or} \qquad 1.8\% \text{ reduction in velocity}$$

$$HP_{\theta} = .949 \ HP_{o} \qquad \text{or} \qquad 5.0\% \text{ reduction in horsepower}$$

(It should be recalled from Eq. (15), however, that for $\theta = 0$ increasing the L/D from 2.5 to 5 results in a 50% reduction in horsepower.)

Figure 24b may also be used to provide the angle of climb resulting from a given increase in thrust at various thrust angles. For example, by confining our attention to the L/D = 3 and L/D = 2.5 curves we note that a difference in flight angle is 3.37^{o}. Now if the flyer has an L/D of 3 but the Parafoil is actually back at an angle equivalent to L/D 2.5 then the angle of climb is 3.37^{o}. Thus by using Figure 24b you can read off the additional horsepower required to obtain this angle of climb at different thrust angles.

TAKE-OFF DISTANCE

Aircraft Case

The conventional aircraft normally operates off a concrete runway and is configured on its landing gear so as to present a minimum drag configuration with no lift. The Take-Off distance therefore is normally estimated by simply computing the distance it takes to accelerate the mass of the aircraft to a velocity of approximately 1.2 times the flight speed. When the aircraft reaches this Take-Off speed the pilot rotates the aircraft to a flight angle of attack and lift-off occurs.

The basic Take-Off equation therefore is given by

$$F = m\,\frac{dv}{dt} = \frac{w}{g}\cdot\frac{dv}{dx}\,\frac{dx}{dt} = \frac{wv}{g}\,\frac{dv}{dx} \tag{20}$$

and the Take-Off distance, X , is obtained by integration of Equation (20) as:

$$X = \frac{W}{g}\int_{o}^{V}\frac{v}{F}\,dv \tag{21}$$

When F is a constant, the Take-Off distance is given by,

$$X = \frac{W}{2g}\;\frac{V^2}{F} \tag{22}$$

If it is assumed that F is the static thrust, (F=T), and that $V = 1.2\ V_S$, then Equation (22) becomes

$$X = \frac{1}{2g}\;\frac{V^2}{(T/W)} \tag{23}$$

$$X = \frac{1.44}{2g}\;\frac{V_S^2}{(T/W)} \tag{24}$$

$$X = \frac{1.44}{2g}\;\frac{(2)}{\rho C_L}\;\frac{(W/A)}{(T/W)} = 18.8\;\frac{(W/A)}{C_L\,(T/W)} \tag{25*}$$

where

$$V_S = \frac{W}{A}\;\frac{2}{\rho C_{L_{max}}} = \text{Stall Velocity} \tag{26}$$

---

*Computed for sea level conditions at $\theta = 0$.

Using these equations the Take-Off distance for the Irish Flyer is given in Figures 25-29 for various values of wing loading, lift coefficient, and thrust loading, (T/W). (During Take-Off the Parafoil is assumed to be overhead and inflated.)

Irish Flyer Case

While the previous analysis may yield good approximate values for the Take-Off distance of an aircraft, the assumption of (1) very large thrust, (2) low drag because of small angle of attack during Take-Off run, and (3) low ground resistance due to concrete runway are poor in the case of the Irish Flyer. The Irish Flyer moves along the ground and in the air at a constant angle of attack. As a result it always experiences lift and drag. As a result thrust is not large as compared with the drag. Further, since the Parafoil normally flys off grass runways the ground resistance is significant and should be considered. As a result in considering the Take-Off distance for the Irish Flyer it is necessary to include the effects of air drag, ground resistance, and static thrust fall off with velocity. Accordingly for the Irish Flyer we may write the total force as:

$$F = T - D - R \tag{27}$$

where

$$T = \underline{\eta}\, T_o \tag{28}$$

$$D = \underline{\eta} T_o (.7^2) = \underline{\eta} T_o \frac{(.7V_{T_o})^2}{{V_{T_o}}^2} \quad \text{(Drag at 70\% of V)}^* \tag{29}$$

$$R = \mu\,(W-L) \qquad \text{(Resistance of 70\% of V)}^* \tag{30}$$

$$= \mu \left[\underline{\eta} T_o \frac{L}{D} - \frac{L}{D} \underline{\eta}\, T_o (.7^2)\right] \tag{31}$$

For flight efficiency a propeller is designed for the best L/D of each blade element at the design flight velocity. At lower flight velocities some of the blade elements may be stalled. At higher speeds the L/D is reduced.** Thus, the realistic Irish Flyer Take-Off distance is given by

---

*Found both experimentally and by analysis to be representative.

**Thrust reduction to increasing flight velocity is due to a decrease in local blade angle of attack.

$$X_T = \frac{w}{2g} \frac{V^2}{(T-D-R)} \quad \text{where } (T\text{-}D\text{-}R) \text{ at } .7V \tag{32}$$

$$= \frac{w}{2g} \frac{V^2}{\eta T_o - \eta T_o (.7^2) - \mu\left[\eta T_o \left(\frac{L}{D}\right)\left(\frac{L}{D}\right) \eta T_o \ (.7^2)\right]} \tag{33}$$

The ratio of Equation (32) and Equation (23) is given by:

$$\frac{X_T}{X} = \frac{\frac{w}{2g} \frac{V^2}{(T-D-R)}}{\frac{w}{2g} \frac{V^2}{T_o}} = \frac{T_o}{T-D-R} \tag{34}$$

$$\frac{X_T}{X} = \frac{T_o}{\underset{\text{True Thrust}}{\eta T_o} - \underset{\text{Drag}}{.49 \eta T_o} - \underset{\text{Ground Resistance}}{\eta T_o \, \mu \left(\frac{L}{D}\right)(.51)}} \tag{35a}$$

$$= \frac{2}{\eta \, (1 - \mu \, {}^{L}/D)} \tag{35b}$$

Equation (35), therefore, provides the ratio of the true Irish Flyer Take-Off distance which includes the effects of ground resistance, air drag, and effective thrust to the Irish Flyer Take-Off distance calculated by considering only the acceleration of the mass of the vehicle to a velocity 20 percent higher than the stall speed.

Figure 30 provides seven examples. Case 1 represents the simple acceleration of the mass of the flyer when neither drag or ground resistance is considered. Case 2 represents the effect of 80% prop efficiency, where it is seen that the Take-Off distance is increased by 25%. Case 3 represents an example of the effect of air drag. Here it is noted that the Take-Off distance is doubled. Case 4 represents the introduction of air drag and 80% prop efficiency. Case 5 represents the effect of ground resistance for a flyer with a lift-to-drag ratio of 3. Case 6 represents the combined effects of air drag and ground resistance for an 80% efficient prop and a lift-to-drag ratio of 3. Case 7 represents the effect of air drag and ground resistance, here the lift-to-drag ratio is only 2. Figure 30 also provides in Cases 6 and 7 values as obtained from Equation 35a.

It is seen in Figure 30 that the true Take-Off distance may be as much as 3.5 times greater than the simple acceleration case. It should, however, be emphasized that, in general, the true Take-Off distances for the Irish Flyer are extremely short as compared to a conventional airplane due primarily to its much smaller wing loading and flight velocity.

PRE-DESIGN ESTIMATES

The curves and equations in the preceding section provide a ready means for estimating the pre-design flight performance of various Irish Flyer designs ranging from 10 pounds to 10,000 pounds. The designer may select any Parafoil or vehicle design he wishes, and then use the basic aerodynamic data to provide conservative or advanced values for the lift and drag coefficients. The various performance curves then yield values for the level flight velocity, the level flight horsepower and the climbing horsepower, as associated with various desired rates of climb or angles of climb. In the following paragraphs some illustrative examples are given as a guide.

Simple Manned Flight (Conservative)

Consider the flight vehicle shown in Figure 31. Its total weight with pilot is 540 pounds and it uses a Parafoil of an aspect ratio of two and a wing area of 360 square feet. The wing loading therefore is, W/A = 1.5.

If a conservative angle of trim of $\alpha = 14^{o}$ is employed, Figures 1 and 5 provide a lift coefficient of $C_L = .75$. The basic drag coefficient is $C_D = .258$. However, as a conservative estimate we will add $\Delta C_L = .076$ to account for additional vehicle drag. As a result, a total drag coefficient of $C_D = .334$ will be used. Thus, the lift-to-drag ratio is 2.2. The level flight velocity may now be obtained from Figure 7 as 41.01 ft/sec or 27.96 miles per hour. The level flight horsepower may be obtained from Figure 11 as HP = 18.

It is seen from Figure 20 that an additional horsepower of HP = 6.6 is required for a rate of climb of 400 ft/min. An angle of climb of $\gamma = 6.5$ is obtained from Figure 22.

Thus, we have found that 25 horsepower should provide a quite conservative flight vehicle performance.

Simple Manned Flight (Advanced)

Again using the basic vehicle of Figure 31 (W=540 pounds) but by cleaning up the aerodynamic design and by using a larger Parafoil (A=400) at a smaller angle of trim ($\alpha = 4^{o}$), we may obtain very significant performance improvement. The aerodynamic data, using Figures 1 and 5 for a Parafoil of an aspect ratio of 3.0, yields a lift coefficient of $C_L = .75$ and a drag coefficient of $C_D = .148$, ($\Delta C_D = 0$). The resulting lift-to-drag ratio is L/D = 5.1. For the wing loading of W/A = 1.5, Figure 7 yields a level

flight velocity of 41.01 ft/sec or 27.96 mph. at a required horsepower of only HP = 8.0 (Figure 11). For a rate of climb of 400 ft/min $\Delta HP=6.6$ is required. Thus, the total horsepower is only 14.43. This represents a 42.3% reduction from the horsepower required in the previous very conservative case.*

Cargo and Weapon Stand-Off Delivery System

Remotely controlled or homing designs may also be considered since both have been successfully demonstrated. For this example a total vehicle system weight of 10,000 pounds will be used with a Parafoil of $A=1666$ ft^2; thus yielding, a wing loading of $W/A=6$. Again using a Parafoil of AR=3.0 at an angle of trim of $\alpha_T = 4^{\circ}$, we obtain $C_L = .75$, $C_D = .148$, and $L/D = 5.1$.

The flight velocity is obtained from Figure 8 as 82.02 ft/sec or 55.9 mph. The required horsepower is HP = 250 (Figure 18) and a $\Delta HP = 60$ is required for a rate of climb of 200 ft/min (Figure 21). Therefore the total horsepower required is HP= 310 to fly this five ton vehicle.

Maneuvering Decoy and Jammer System

For aircraft drop, a maneuvering decoy-jammer system of 10 pounds is considered which uses a Parafoil having an area of 6.66 square feet and a wing loading of $W/A = 1.5$. Again using an AR 3 Parafoil with a lift coefficient $C_L = .75$ and a drag coefficient of $C_D = .148$ ($\Delta C_D = 0$), we obtain a flight velocity of U = 41.01 ft/sec or 27.96 mph (Figure 7) and a horsepower of HP=.146 (Figure 10). (For this example it is interesting to note that successful flight demonstrations have already been carried out using model aircraft engines and model aircraft control systems.)

*Underwater designs may also be considered. For example an unmanned 540 pound system with a specific gravity of 3 might have a compact underwater weight of 360 pounds. If the 360 ft^2 Parafoil is used, the flight velocity would be only V= 1.217 ft/sec and the required horsepower HP=.156 since the density ratio of water-to-air is approximately 770.

If a smaller Parafoil of 60 ft^2 were used the velocity would be V=2.96 ft/sec or 2.0 mph and the horsepower would be HP = .379. (Some underwater tests have been carried out on gliding systems.

Minimum Manned Vehicle

Various examples of minimum manned vehicles have been carried out and are shown in Figures 32-34. The reader is encouraged to try his hand at the design of a minimum manned vehicle. It should be noted that Parafoils of larger area and lighter weight are readily possible and, also, that improved canopy lift coefficients and lift-to-drag ratios approaching 8 are considered feasible.

CONCLUSIONS*

The Parafoil aerodynamic data and Irish Flyer flight performance curves and equations are presented so as to provide the interested designer with pre-design performance estimates for system designs ranging from 10 pounds to 10,000 pounds. Some examples of manned flight, cargo and weapon delivery, decoy and jammer system, and underwater flight have been set forth so as to aid the designer. Individual designers will of course optimize for their own special application.

*On the 70th anniversary of the Wright Brothers flight, 17 December 1973, Dr. John D. Nicolaides was privileged to carry out a flight demonstration of Irish Flyer N3029 at Goshen, Indiana for the U.S. Air Force representative of the Flight Dynamics Laboratory, Mr. Michael Higgins. The flyer demonstrated climb from 50 ft. to 1,300 ft., right turns, left turns, complete static and dynamic flight stability, complete control, and an accurate soft landing. This flight was documented by Mr. Morley Safer of CBS-TV for nationwide presentation on the "CBS-60 minutes" TV show.

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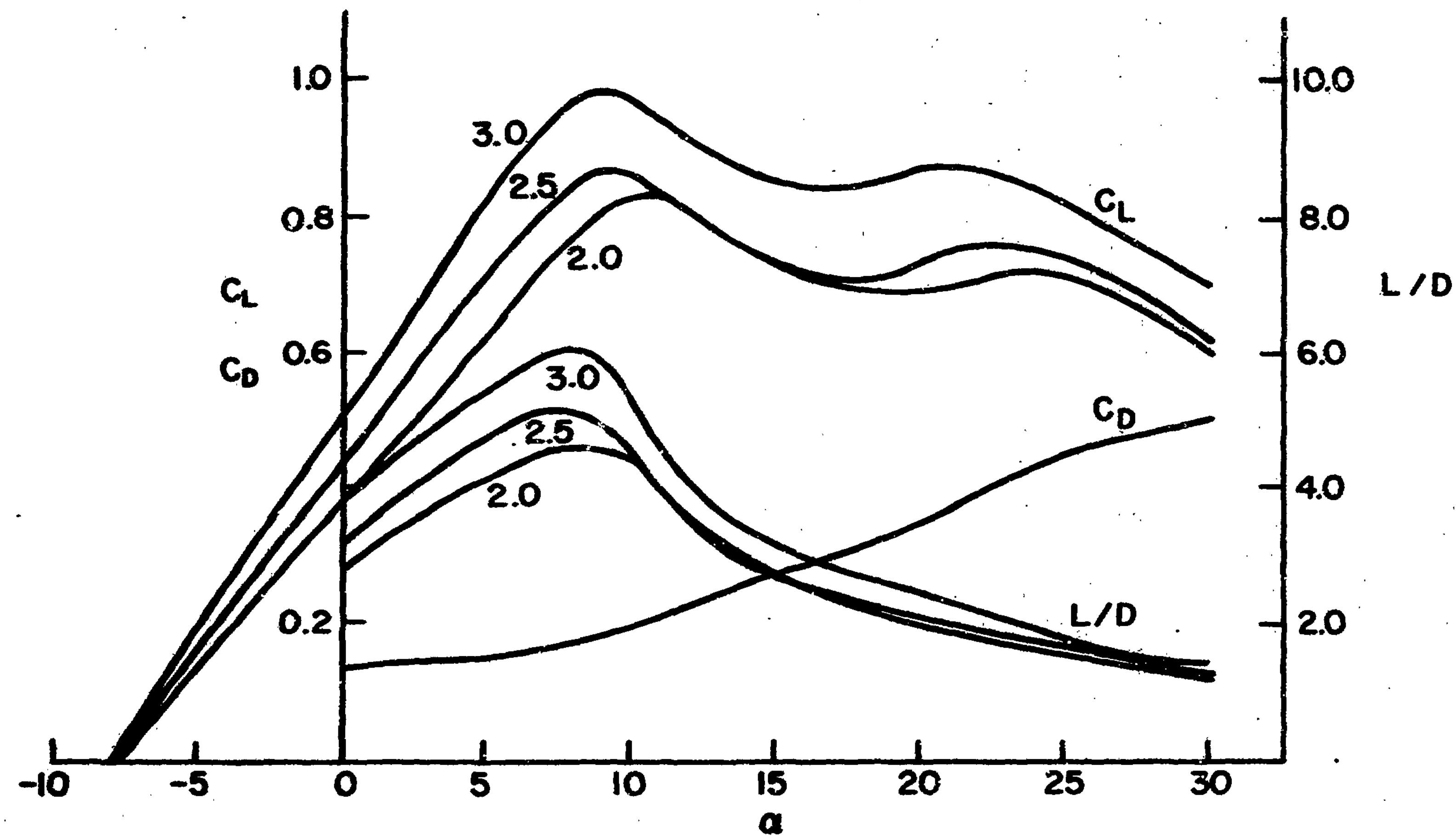

Figure 1. Summary of Aerodynamic Data

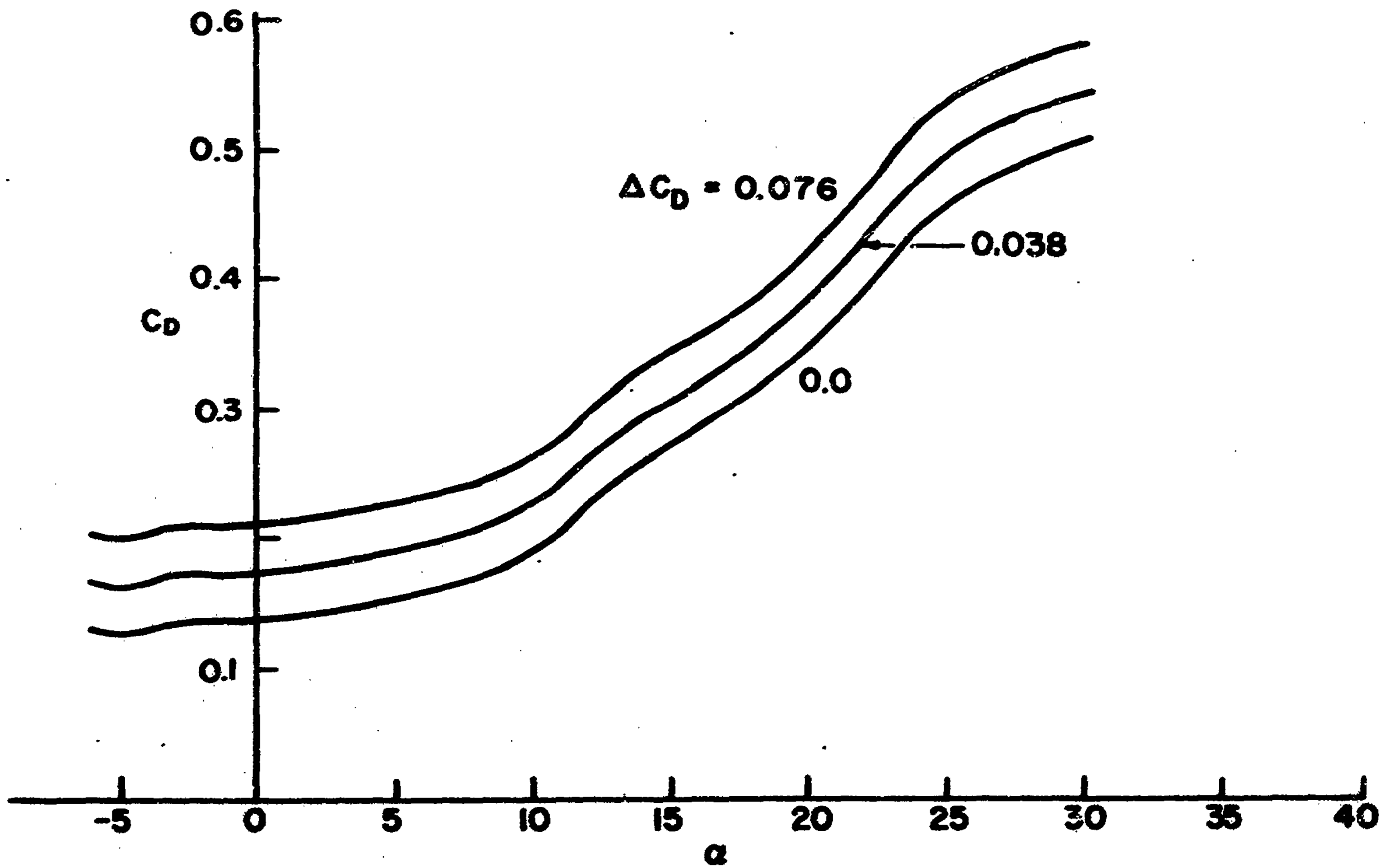

Figure 2. Aerodynamic Drag Data

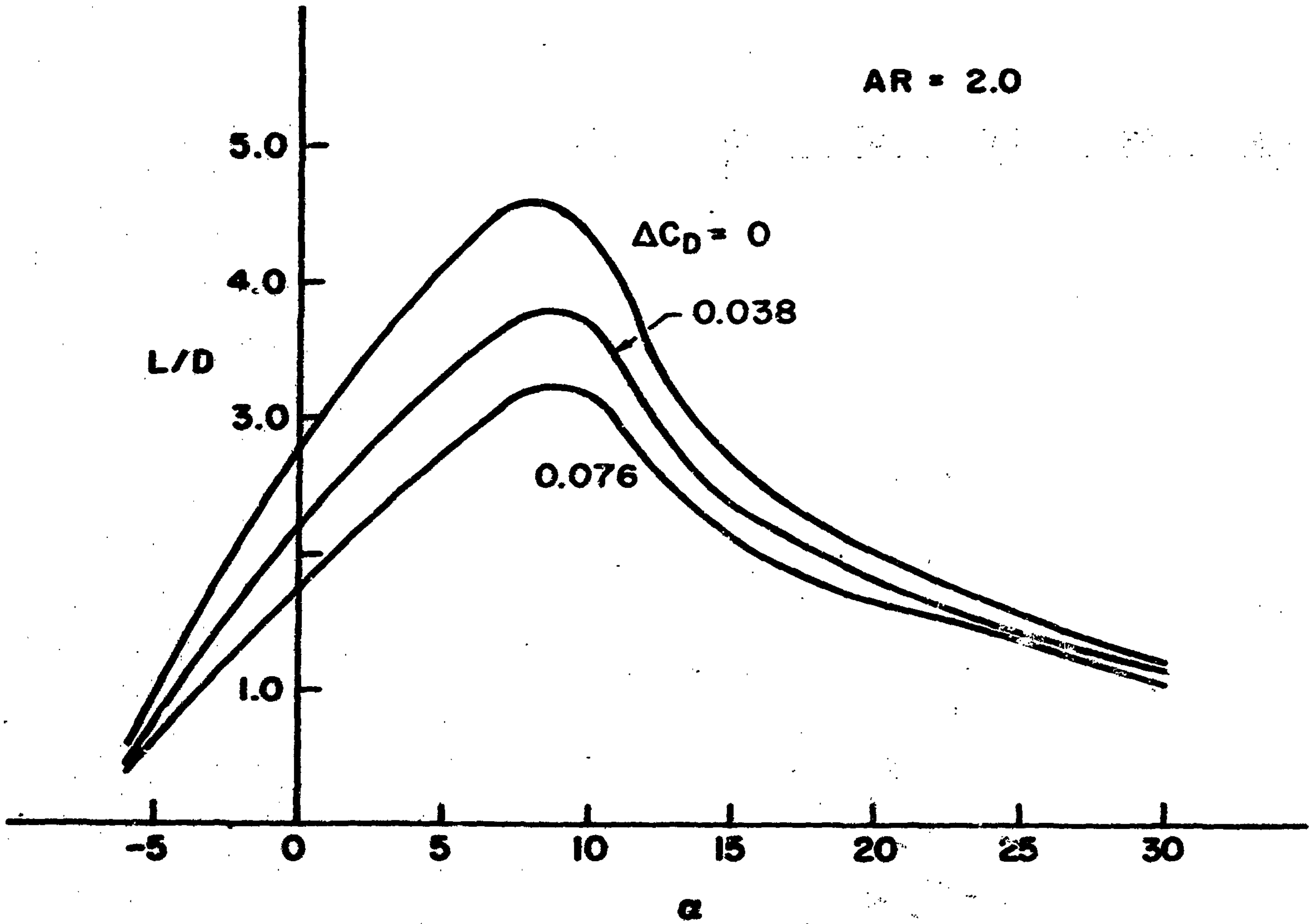

Figure 3. Lift-to-Drag Ratios for ND 2.0 for Three System Drag Increments

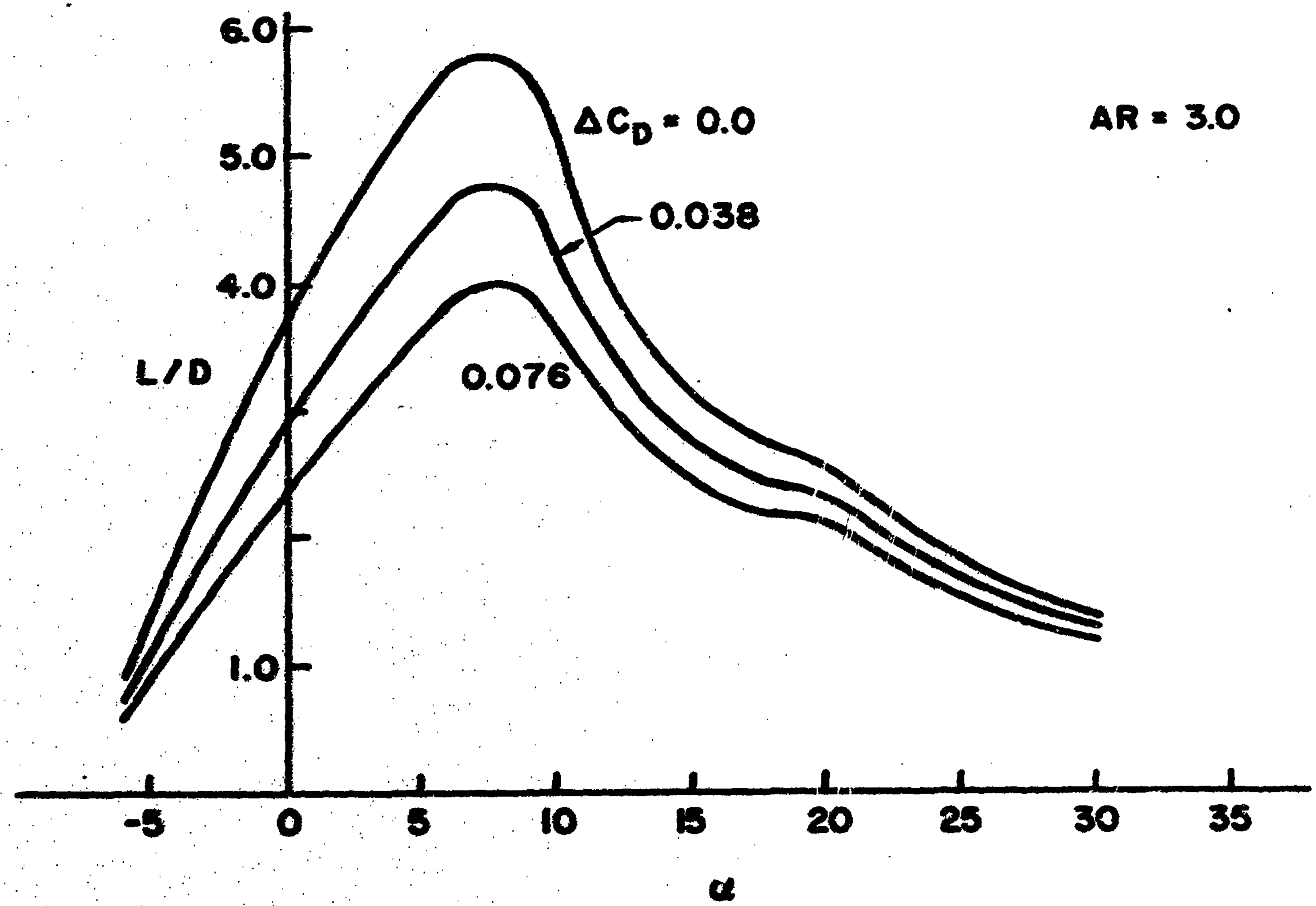

Figure 4. Lift-to-Drag Ratios for ND 3.0 for Three System Drag Increments

α	C_D	$C_D+\Delta C_D$ $\Delta C_D=+.038$	$C_D+\Delta C_D$ $\Delta C_D=+.076$	C_L AR=2.0	L/D AR=2.0 $\Delta C_D=0$	L/D AR=2.0 $\Delta C_D=.038$	L/D AR=2.0 $\Delta C_D=.076$	AR = 2.5 C_L	AR = 2.5 L/D($\Delta C_D=0$)	AR = 3.0 C_L	AR = 3.0 L/D $\Delta C_D=0$	AR = 3.0 L/D $\Delta C_D=.038$	AR = 3.0 L/D $\Delta C_D=.076$
-6	.130	.168	.206	.077	.592	.458	.373	.100	.769	.120	.923	.714	.582
-5	.122	.160	.198	-	.991	.758	.611	-	-	-	-	-	-
-4	.128	.167	.205	.172	1.343	1.029	.839	.208	1.625	.250	1.953	1.497	1.219
-3	.133	.171	.209	-	1.669	1.321	1.081	-	-	-	-	-	-
-2	.132	.170	.208	.276	2.090	1.623	1.320	.320	2.424	.381	2.886	2.241	1.831
-1	.133	.171	.209	-	2.428	1.888	1.545	-	-	-	-	-	-
0	.136	.174	.212	.377	2.772	2.166	1.778	.430	3.161	.500	3.676	2.873	2.358
1	.137	.175	.213	-	3.087	2.417	1.985	-	-	-	-	-	-
2	.142	.180	.218	.477	3.359	2.650	2.188	.542	3.816	.628	4.422	3.488	2.[illegible]80
3	.145	.183	.221	-	3.627	2.874	2.380	-	-	-	-	-	-
4	.148	.186	.224	.576	3.891	3.096	2.571	.657	4.439	.756	5.108	4.064	3.375
5	.152	.190	.228	-	4.092	3.273	2.728	-	-	-	-	-	-
6	.155	.193	.231	.676	4.361	3.502	2.926	.769	4.961	.883	5.696	4.575	3.822
7	.160	.198	.236	-	4.531	3.661	3.072	-	-	-	-	-	-
8	.168	.206	.244	.772	4.595	3.747	3.163	.845	5.029	.967	5.755	4.694	3.963
9	.180	.218	.256	-	4.566	3.770	3.210	-	-	-	-	-	-
10	.187	.225	.263	.828	4.427	3.680	3.148	.860	4.598	.970	5.187	4.311	3.688
11	.204	.242	.280	-	4.049	3.413	2.950	-	-	-	-	-	-
12	.228	.260	.298	.805	3.530	3.096	2.701	.802	3.517	.917	4.021	3.526	3.077
13	.244	.282	.320	-	3.196	2.765	2.617	-	-	-	-	-	-
14	.258	.296	.334	.748	2.918	2.543	2.188	.750	2.906	.868	3.364	2.932	2.598
15	.270	.308	.346	-	2.696	2.363	2.104	-	-	-	-	-	-
16	.284	.322	.360	.711	2.503	2.208	1.975	.718	2.528	.841	2.961	2.611	2.336
17	.294	.332	.370	-	2.357	2.057	1.872	-	-	-	-	-	-
18	.312	.350	.388	.688	2.205	1.965	1.773	.712	2.282	.850	2.724	2.428	2.190
19	.329	.367	.404	-	2.072	1.858	1.683	-	-	-	-	-	-
20	.344	.382	.430	.665	1.991	1.793	1.630	.735	2.136	.887	2.578	2.321	2.111
21	.366	.404	.422	-	1.898	1.720	1.572	-	-	-	-	-	-
22	.392	.430	.468	.710	1.811	1.651	1.517	.759	1.936	.872	2.224	2.027	1.863
23	.415	.453	.491	-	1.734	1.589	1.466	-	-	-	-	-	-
24	.439	.477	.515	.722	1.644	1.513	1.401	.758	1.726	.842	1.917	1.883	1.634
25	.458	.496	.534	-	1.561	1.441	1.338	-	-	-	-	-	-
26	.468	.506	.544	.700	1.495	1.383	1.286	.728	1.549	.790	1.688	1.561	1.452
27	.484	.522	.560	-	1.400	1.298	1.210	-	-	-	-	-	-
28	.485	.523	.561	.645	1.329	1.233	1.149	.679	1.400	.745	1.536	1.424	1.327
29	.495	.533	.571	-	1.262	1.172	1.094	-	-	-	-	-	-
30	.507	.545	.583	.605	1.193	1.110	1.037	.621	1.224	.698	1.396	1.288	1.197

Figure 5. Summary of Aerodynamic Data

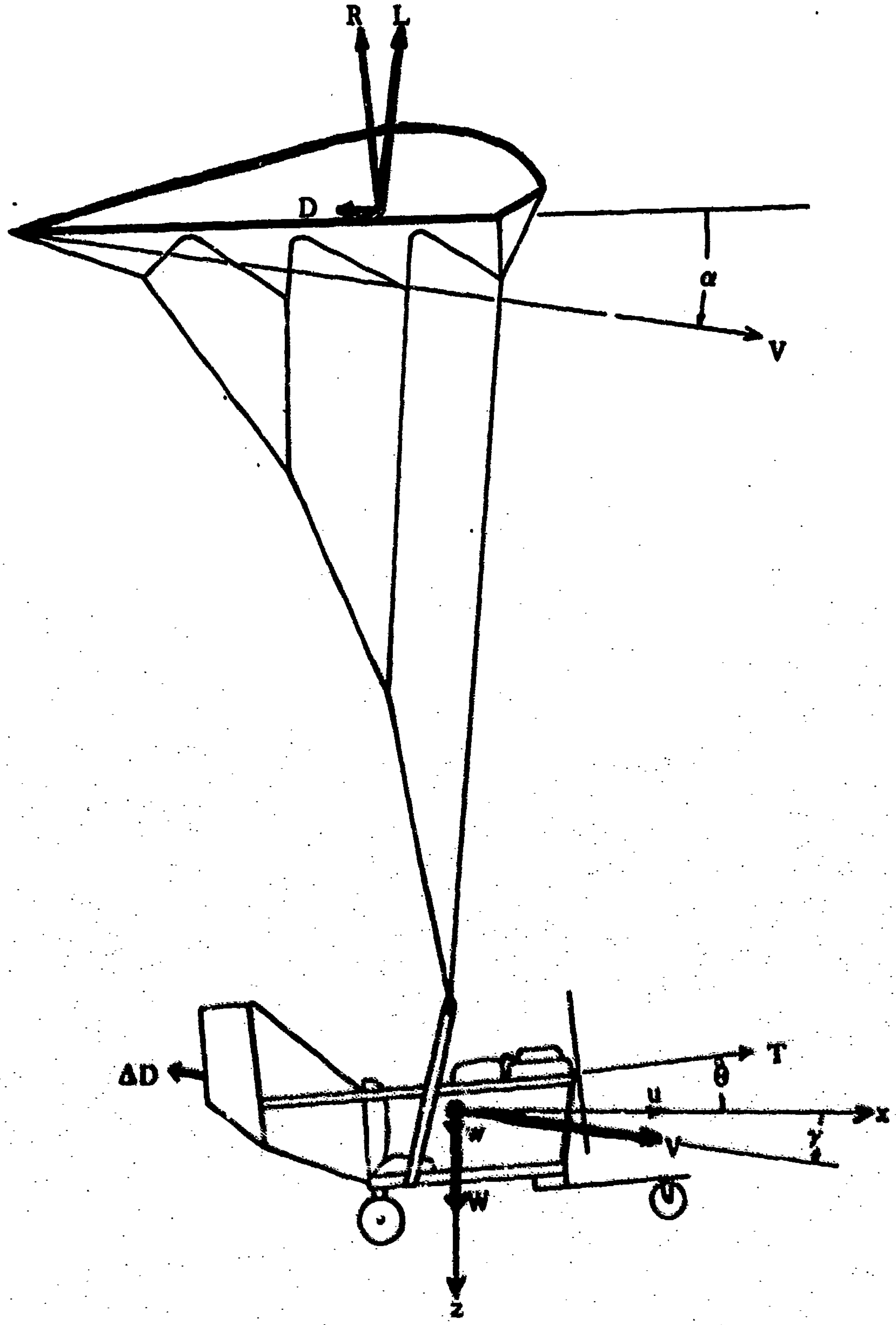

Figure 6. Irish Flyer in Flight

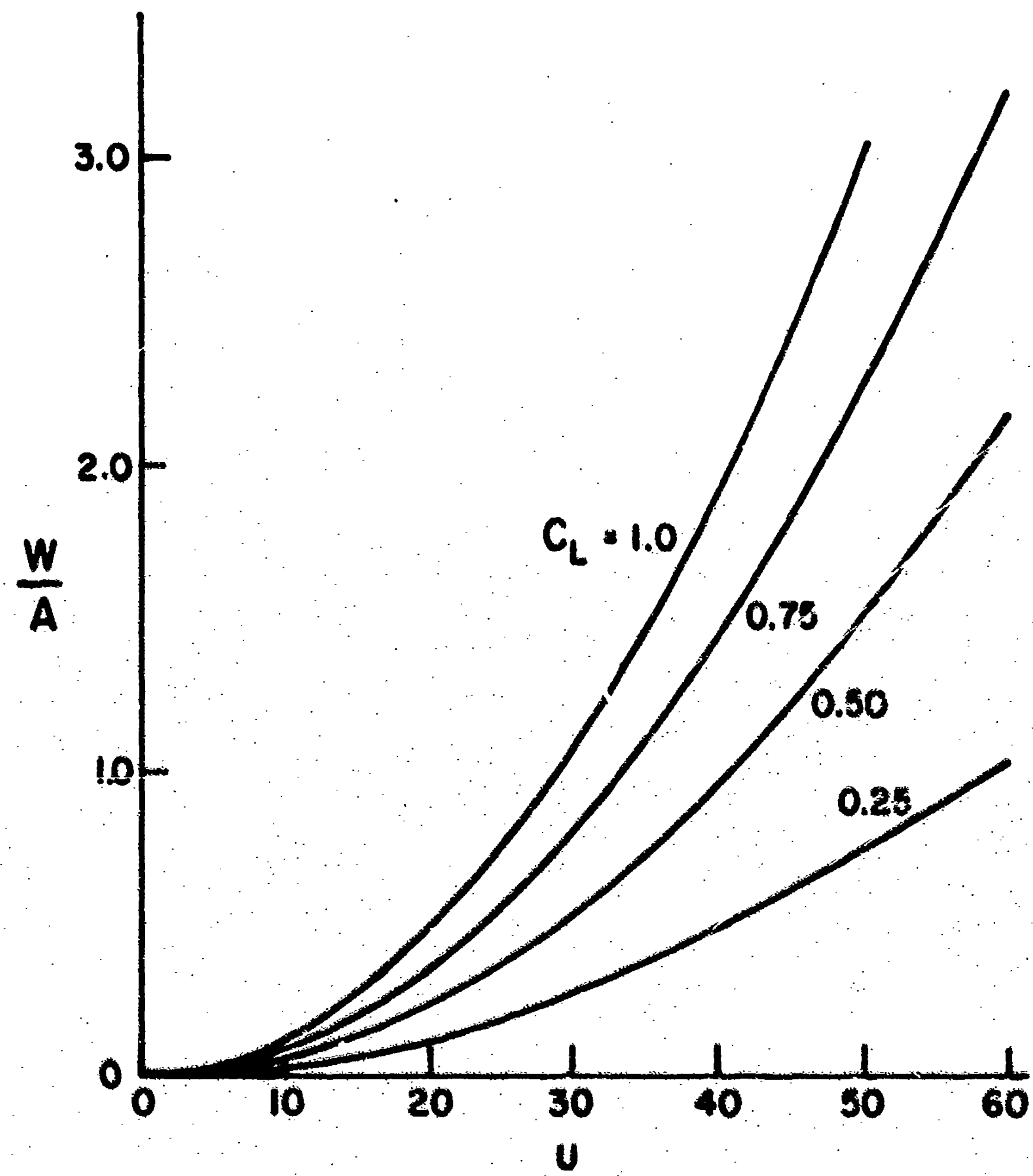

Figure 7. Wing Loading vs Flight Velocity for Various Values of Lift Coefficient ($0 \leq W/A \leq 3$)

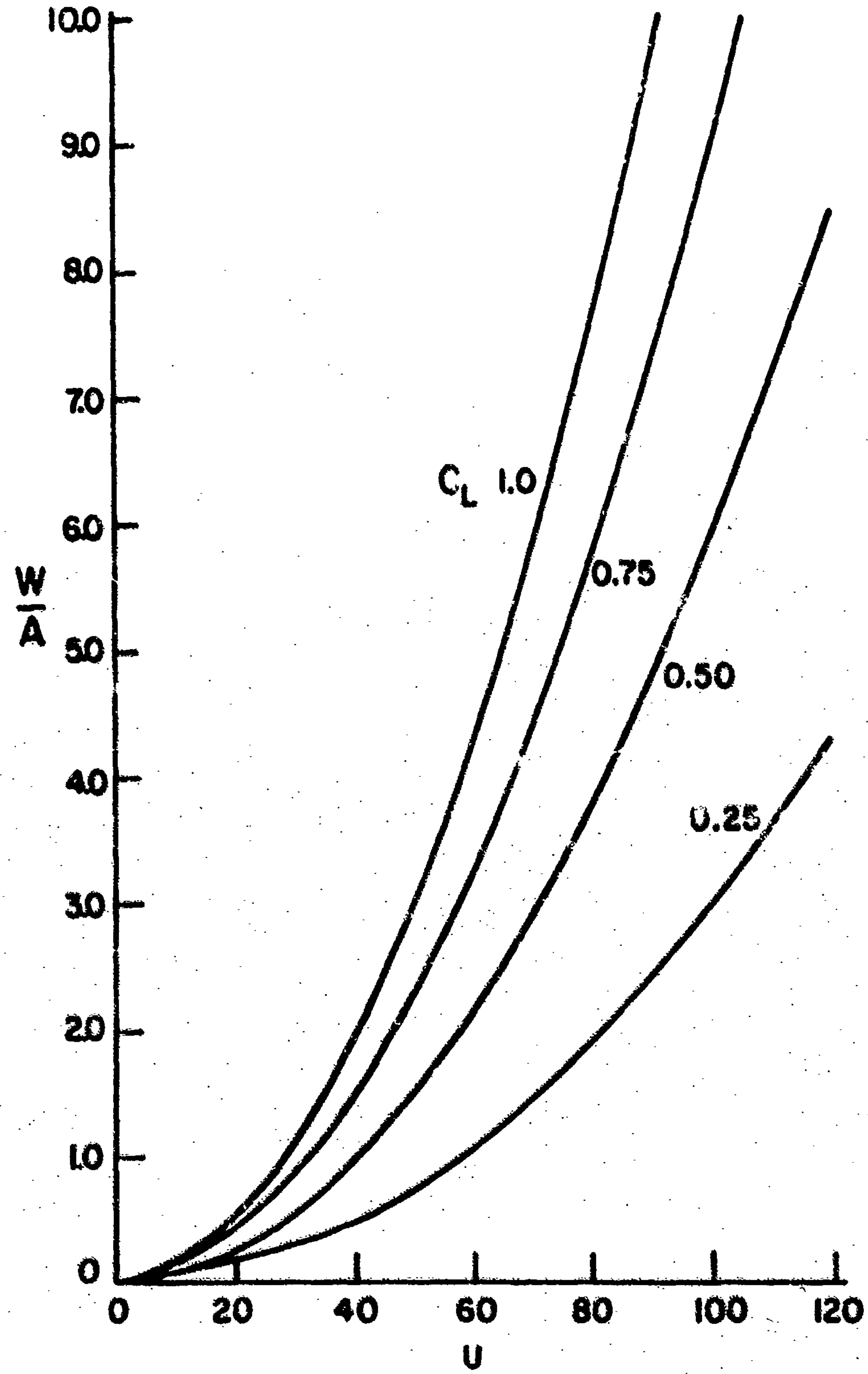

Figure 8. Wing Loading vs Flight Velocity for Various Values of Lift Coefficients ($0 \leq W/A \leq 3$)

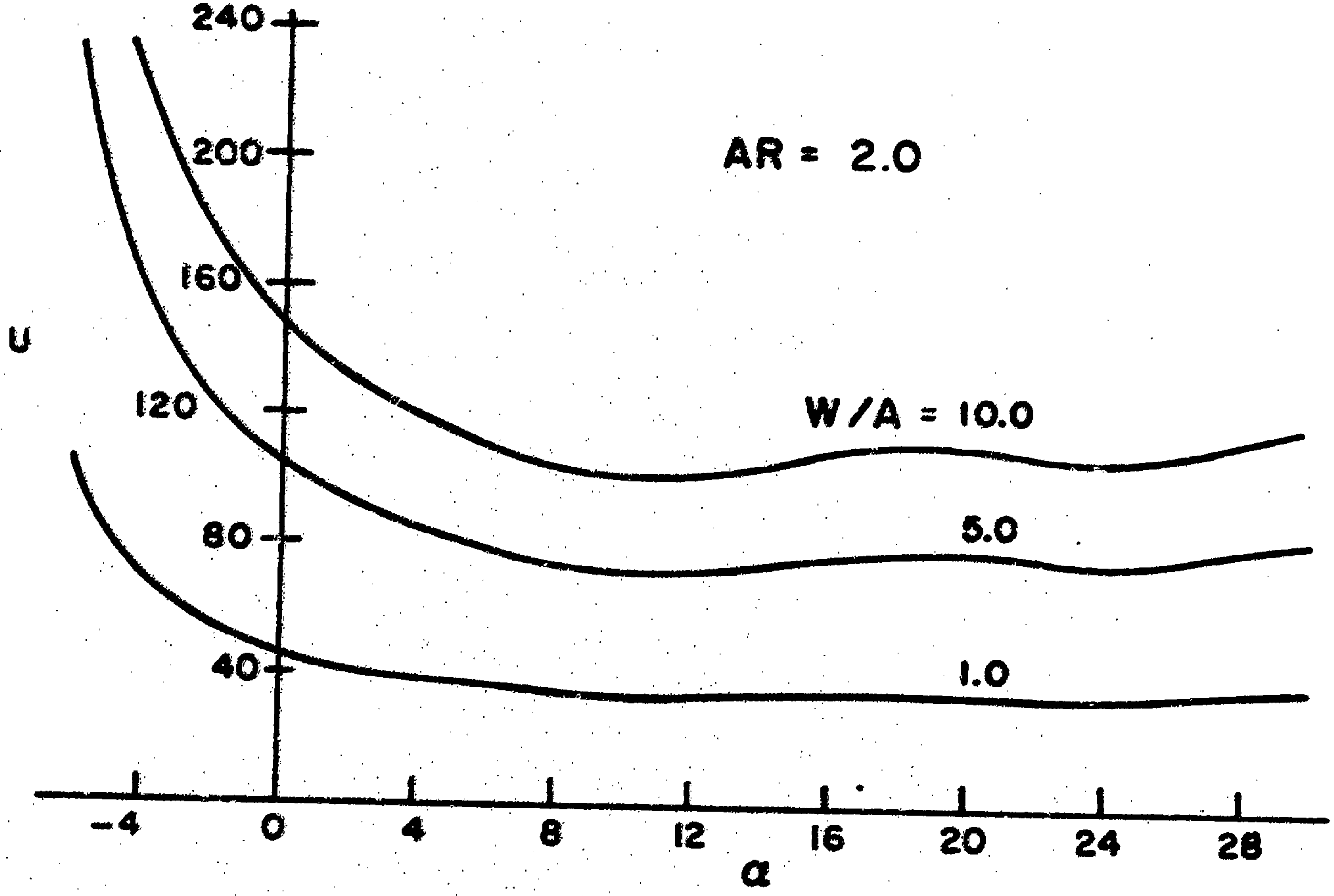

Figure 9. Flight Velocity vs Angle of Attack for W/A of 1.0, 5.0, and 10.0

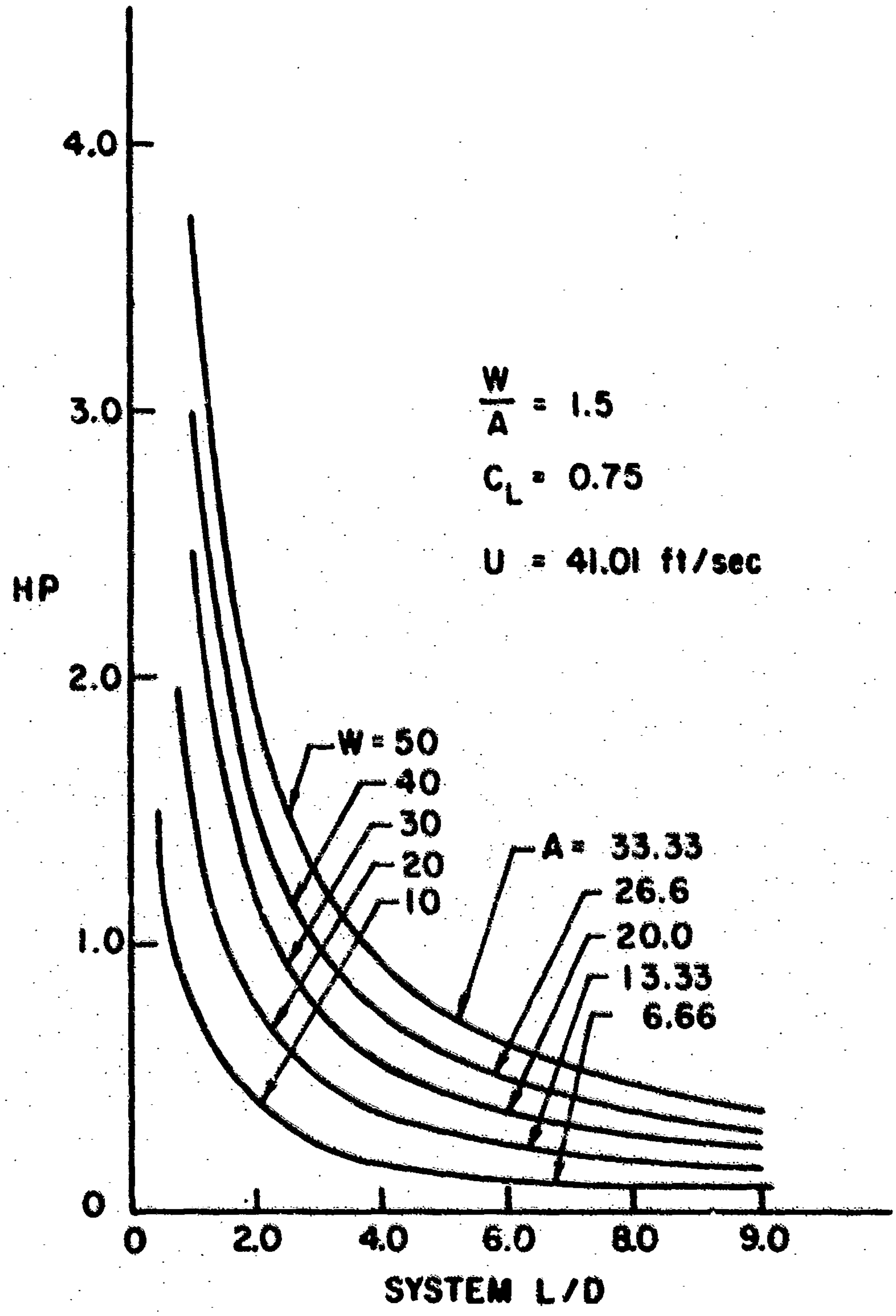

Figure 10. Level Flight Horsepower vs Lift-to-Drag Ratio for W/A = 1.5 and C_L = .75 ($10 \leq W \leq 50$)

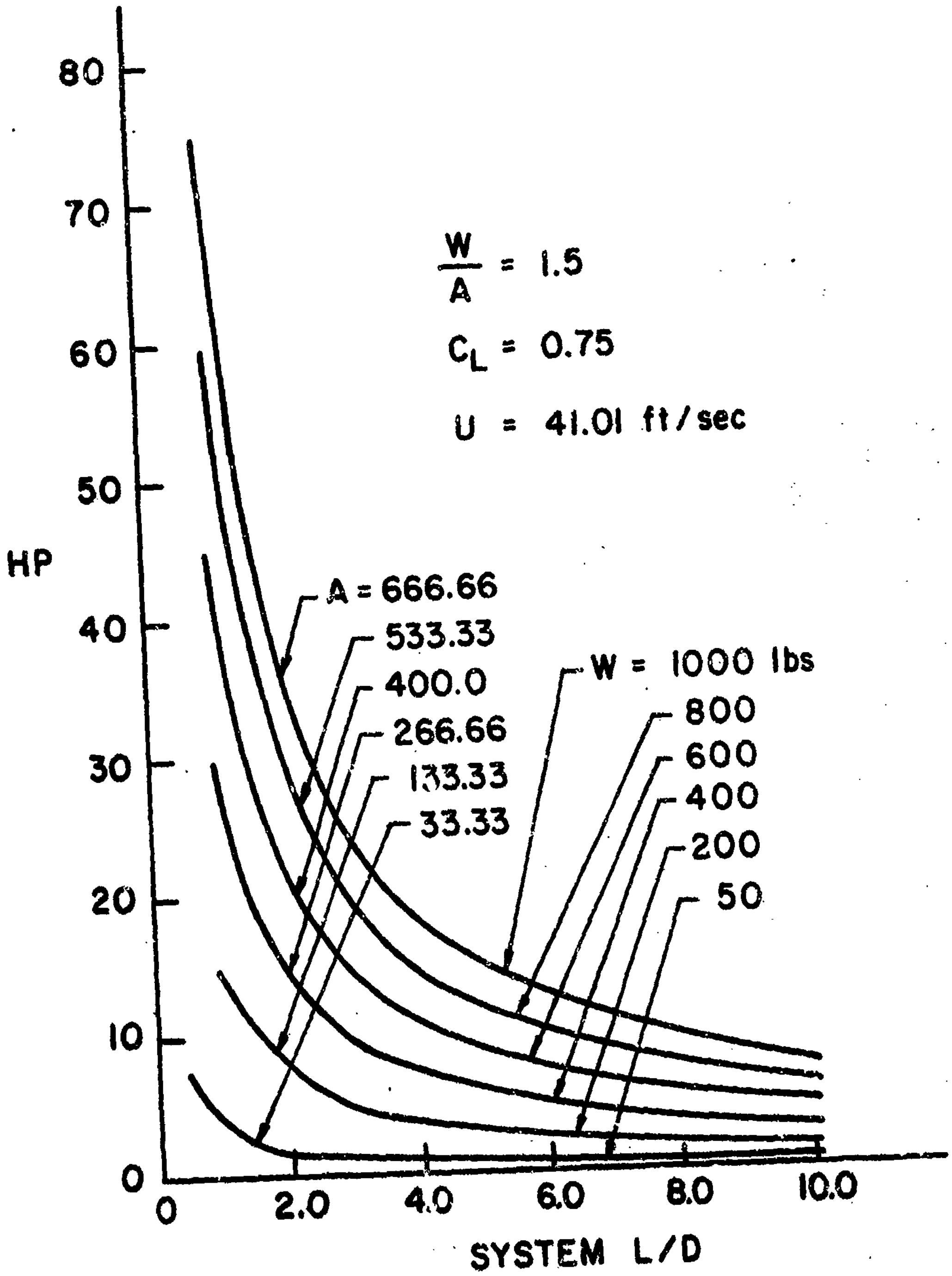

Figure 11. Level Flight Horsepower vs Lift-to-Drag Ratio for $W/A = 1.5$ and $C_L = .75$ ($50 \leq W \leq 1,000$)

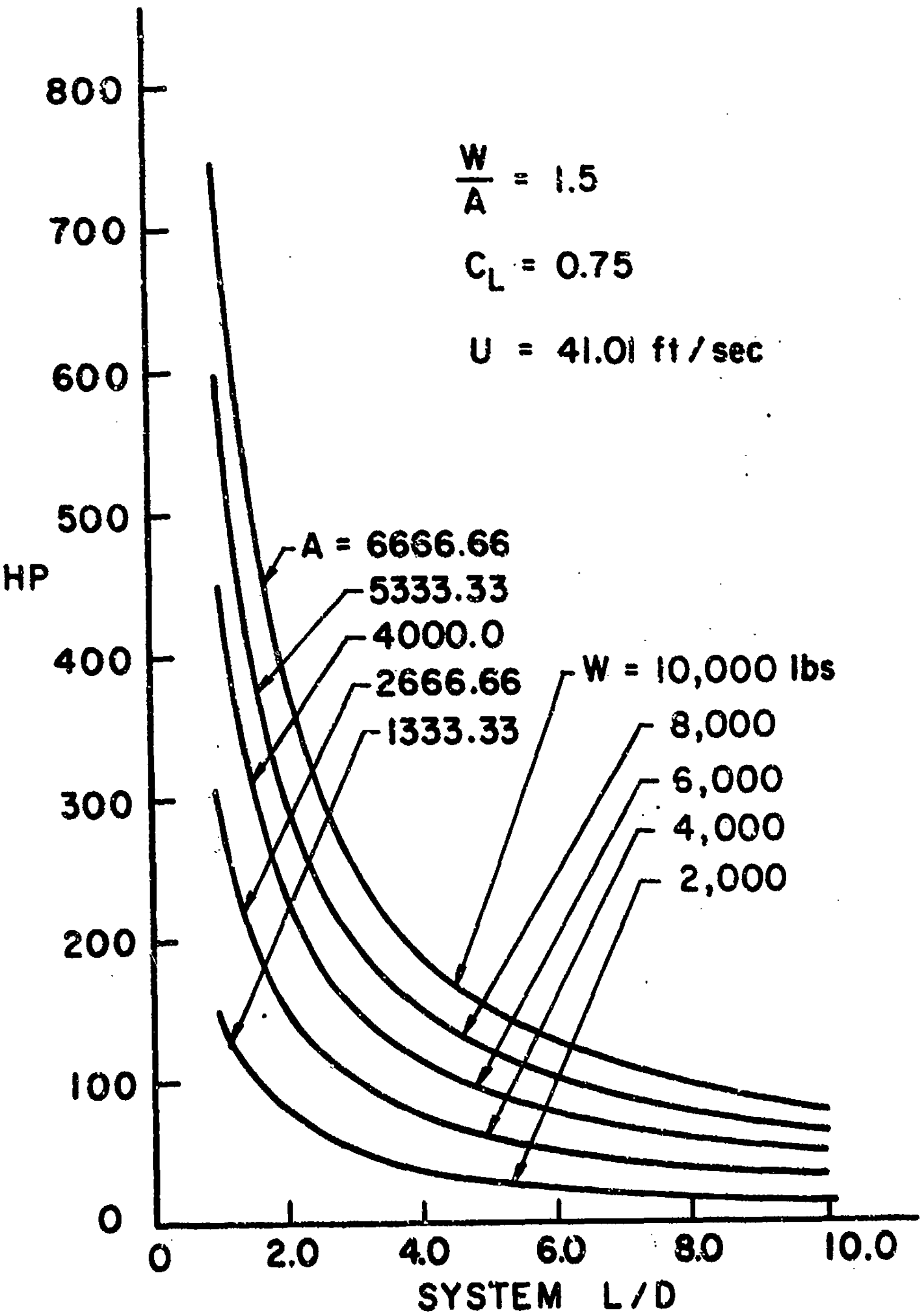

Figure 12. Level Flight Horsepower vs Lift-to-Drag Ratio for $W/A = 1.5$ and $C_L = .75$ $(2000 \leq W \leq 10,000)$

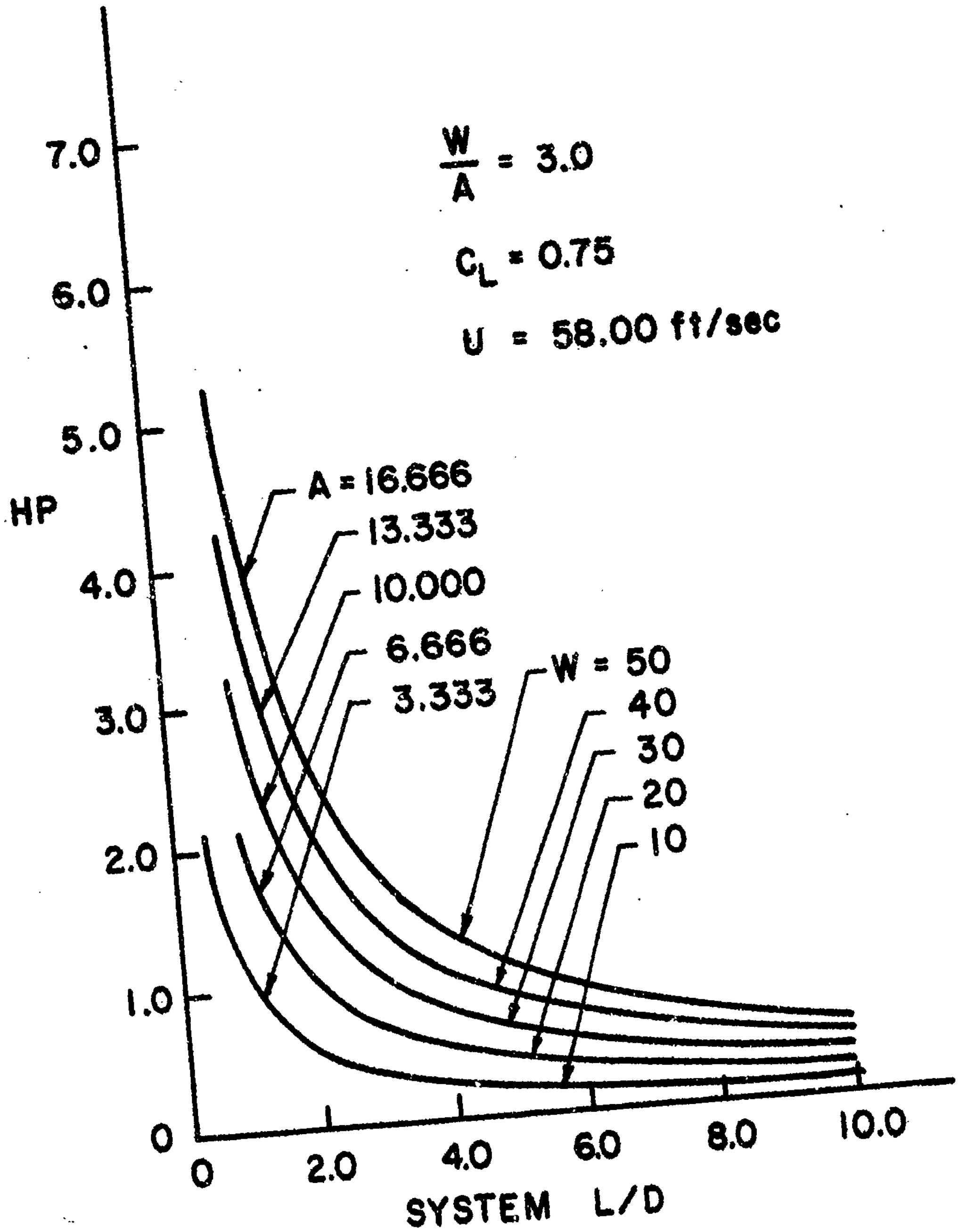

Figure 13. Level Flight Horsepower vs Lift-to-Drag Ratio for W/A = 3.0 and C_L = .75 ($10 \leq W \leq 50$)

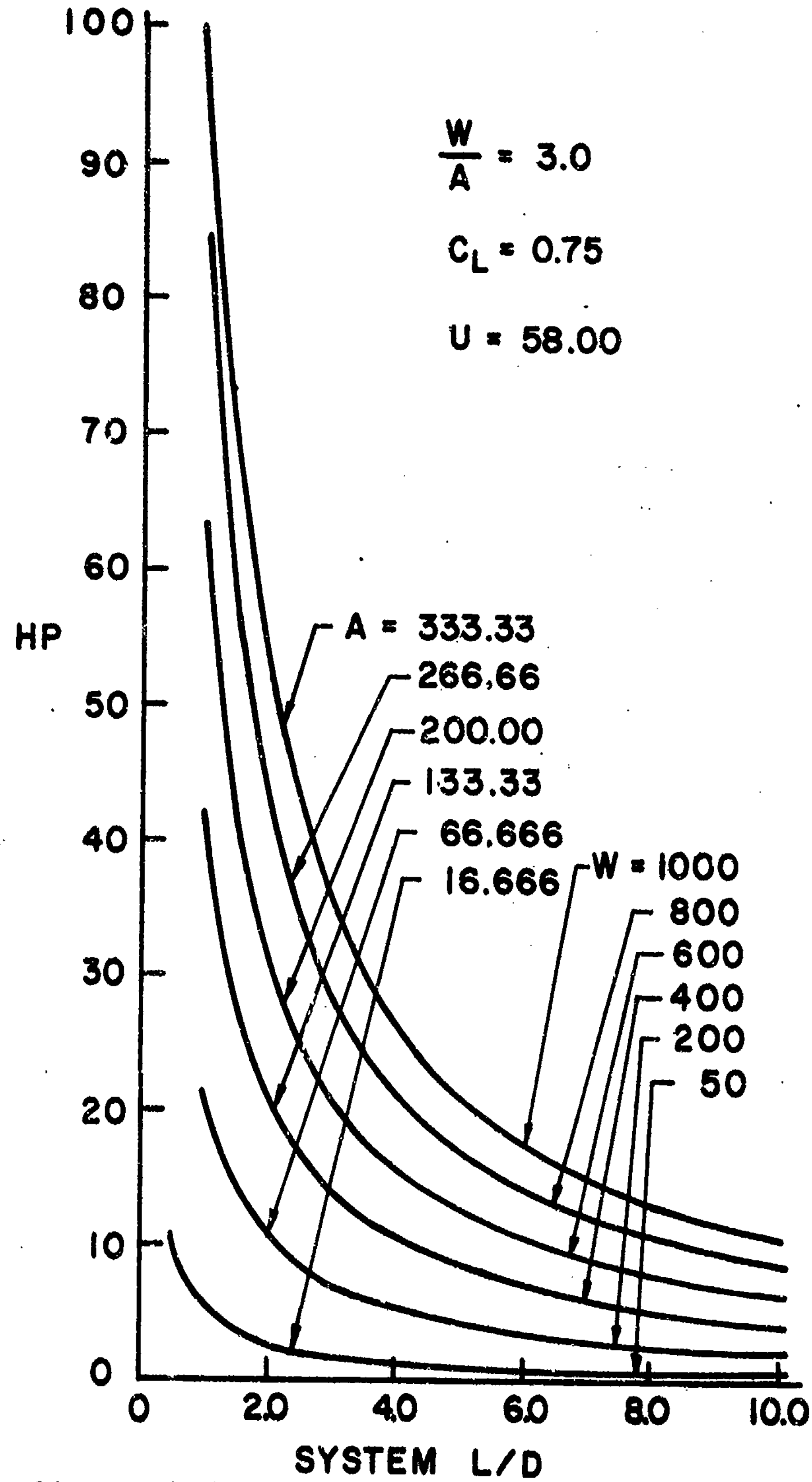

Figure 14. Level Flight Horsepower vs Lift-to-Drag Ratio for W/A = 3.0 and C_L = .75 ($50 \leq W \leq 1,000$)

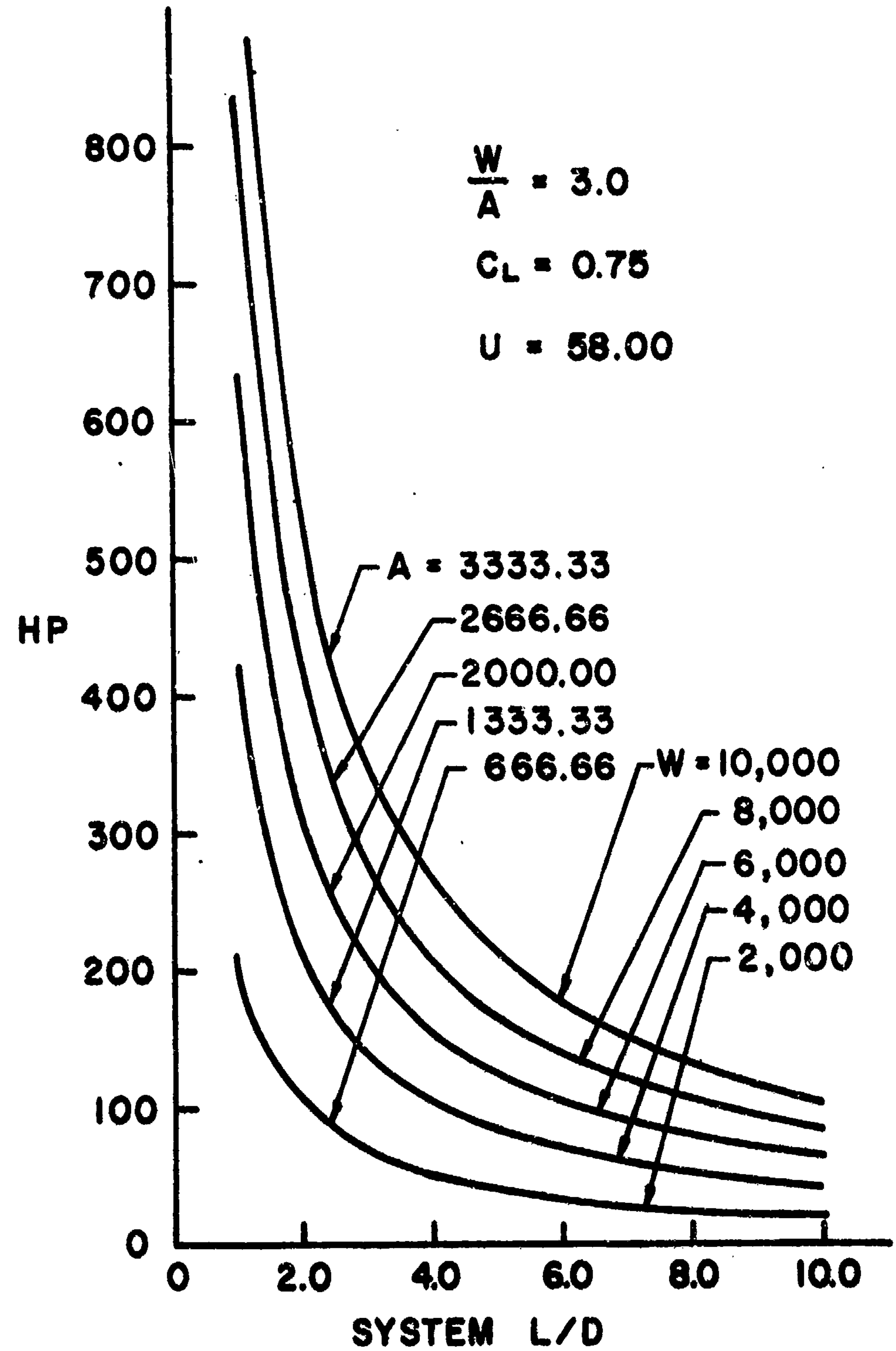

Figure 15. Level Flight Horsepower vs Lift-to-Drag Ratio for W/A = 3.0 and C_L = .75 (2,000 < W < 10,000)

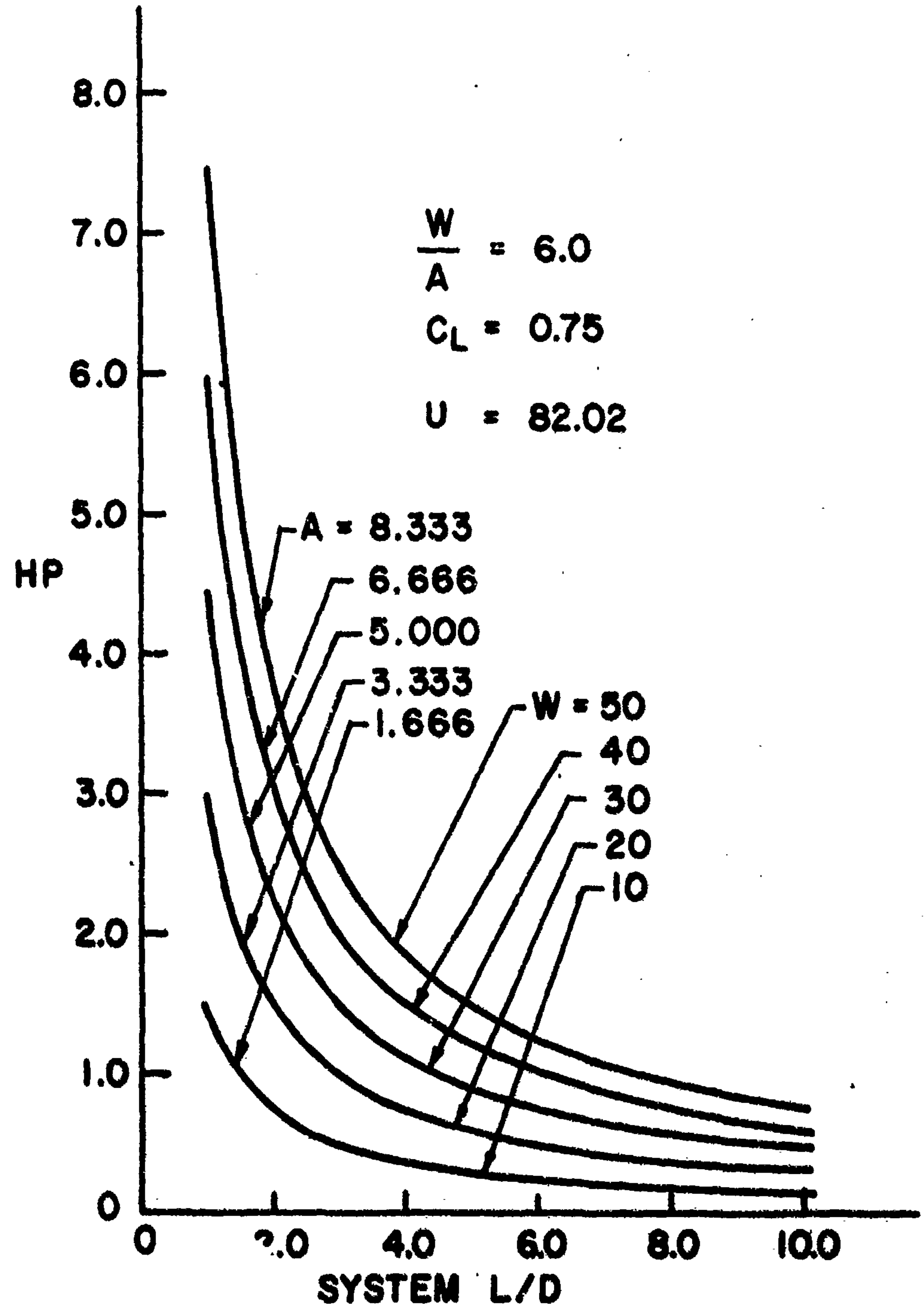

Figure 16. Level Flight Horsepower vs Lift-to-Drag Ratio for $W/A = 6.0$ and $C_L = .75$ $(10 \leq W \leq 50)$

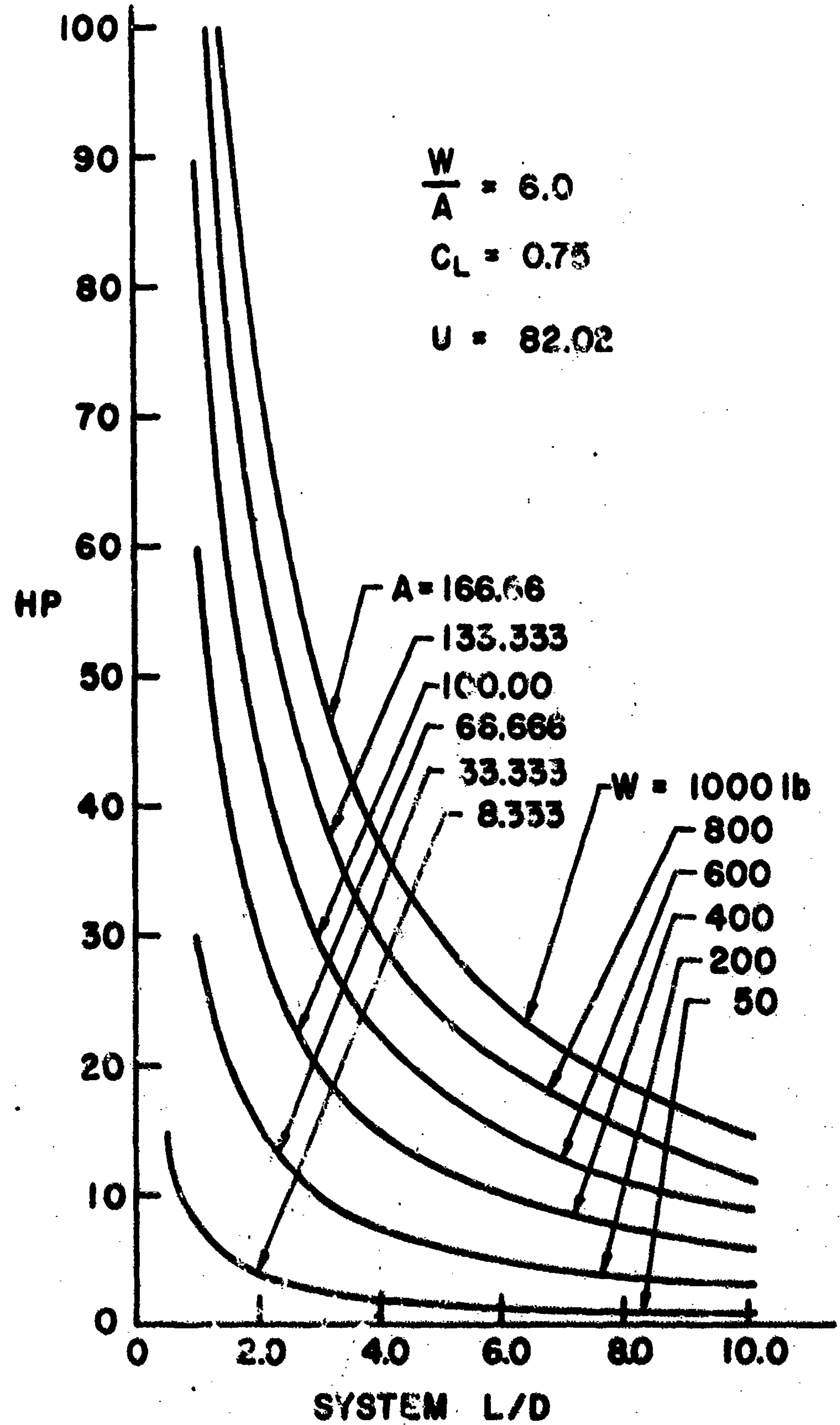

Figure 17. Level Flight Horsepower vs Lift-to-Drag Ratio for W/A = 6.0 and C_L = .75 ($50 \leq W \leq 1{,}000$)

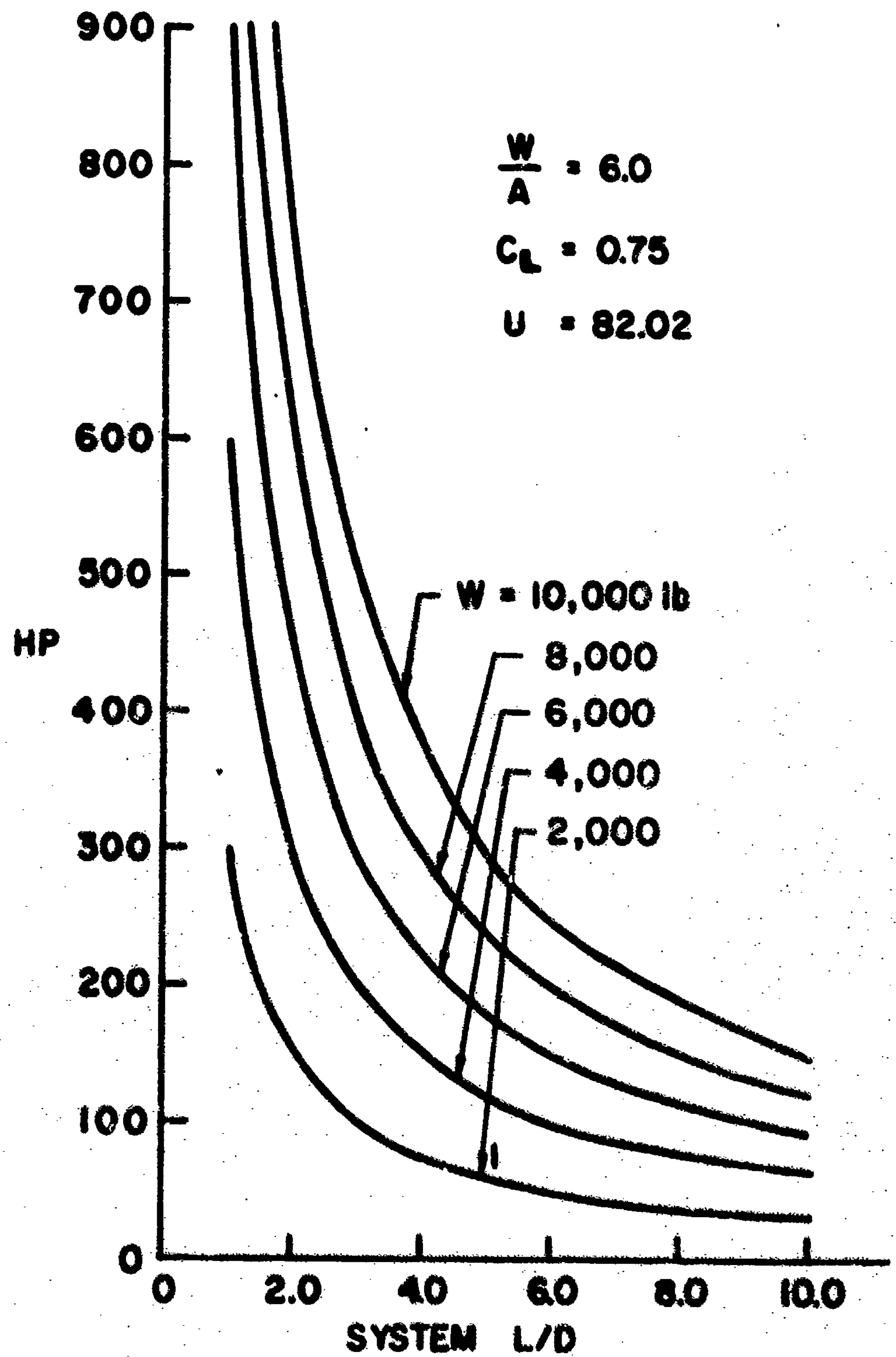

Figure 18. Level Flight Horsepower vs Lift-to-Drag Ratio for $W/A = 6.0$ and $C_L = .75$ ($2{,}000 < W < 10{,}000$)

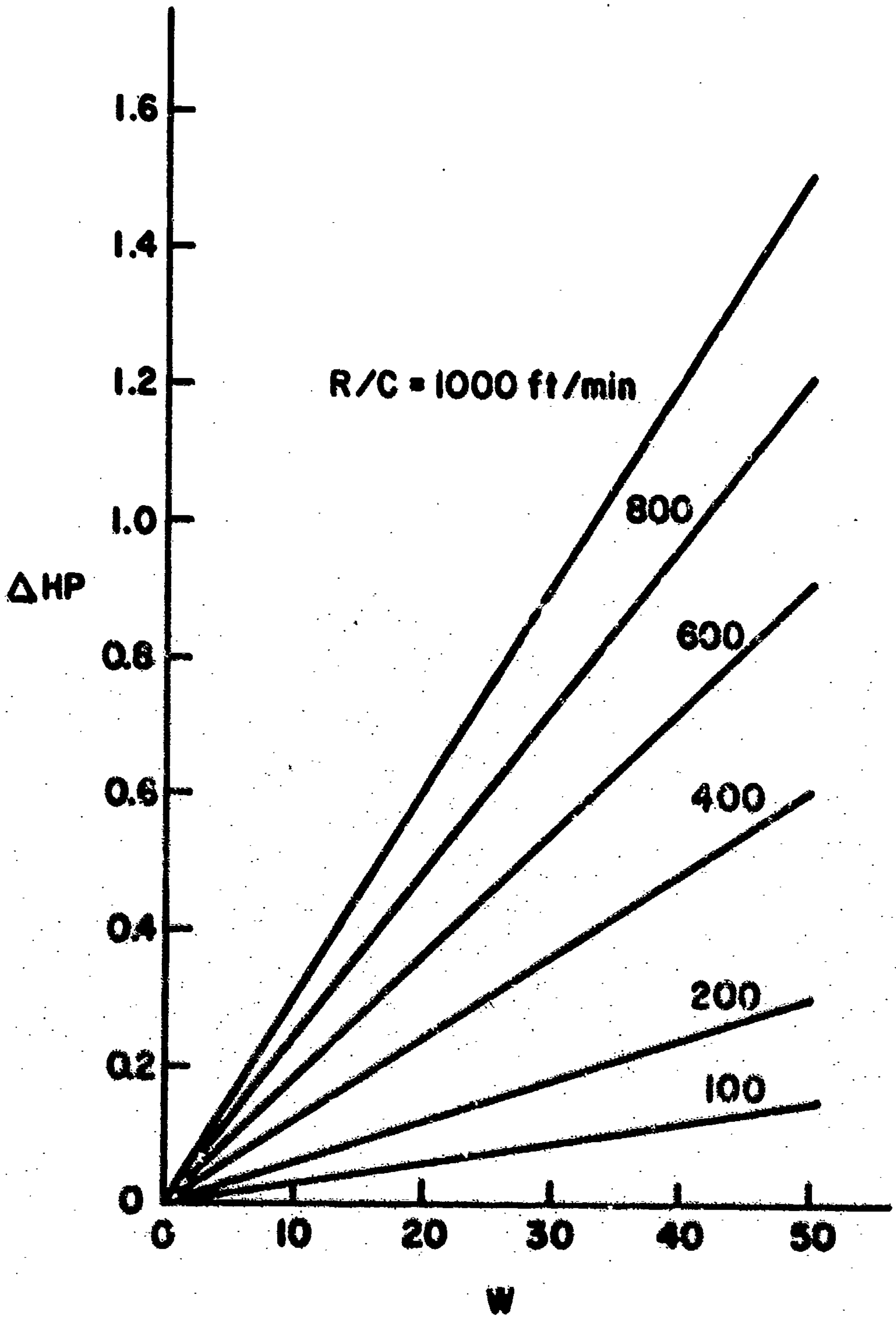

Figure 19. Additional Horsepower vs Weight for Various Rates of Climb $(0 < W < 50)$

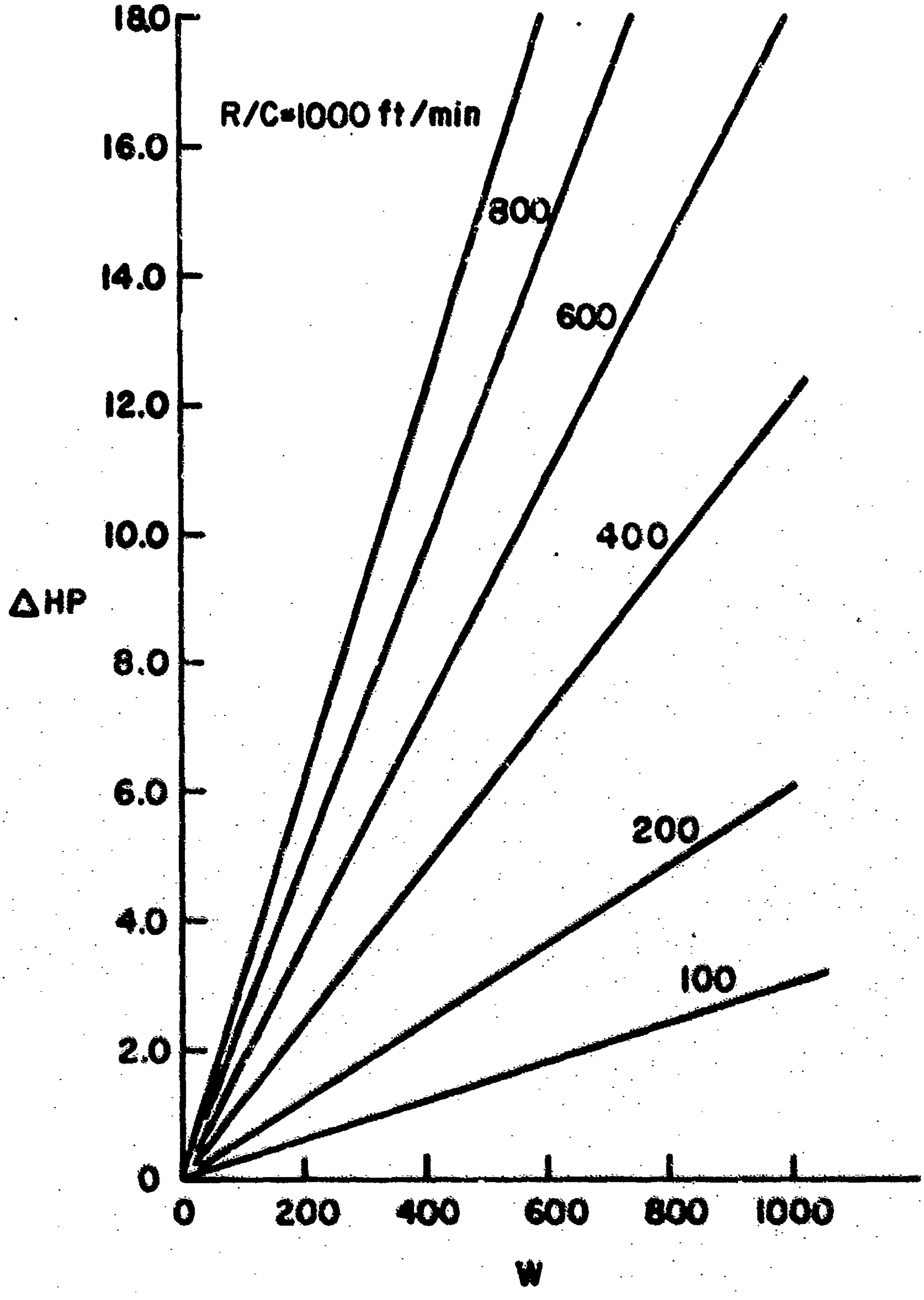

Figure 20. Additional Horsepower vs Weight for Various Rates of Climb ($0 \leq W \leq 1,000$)

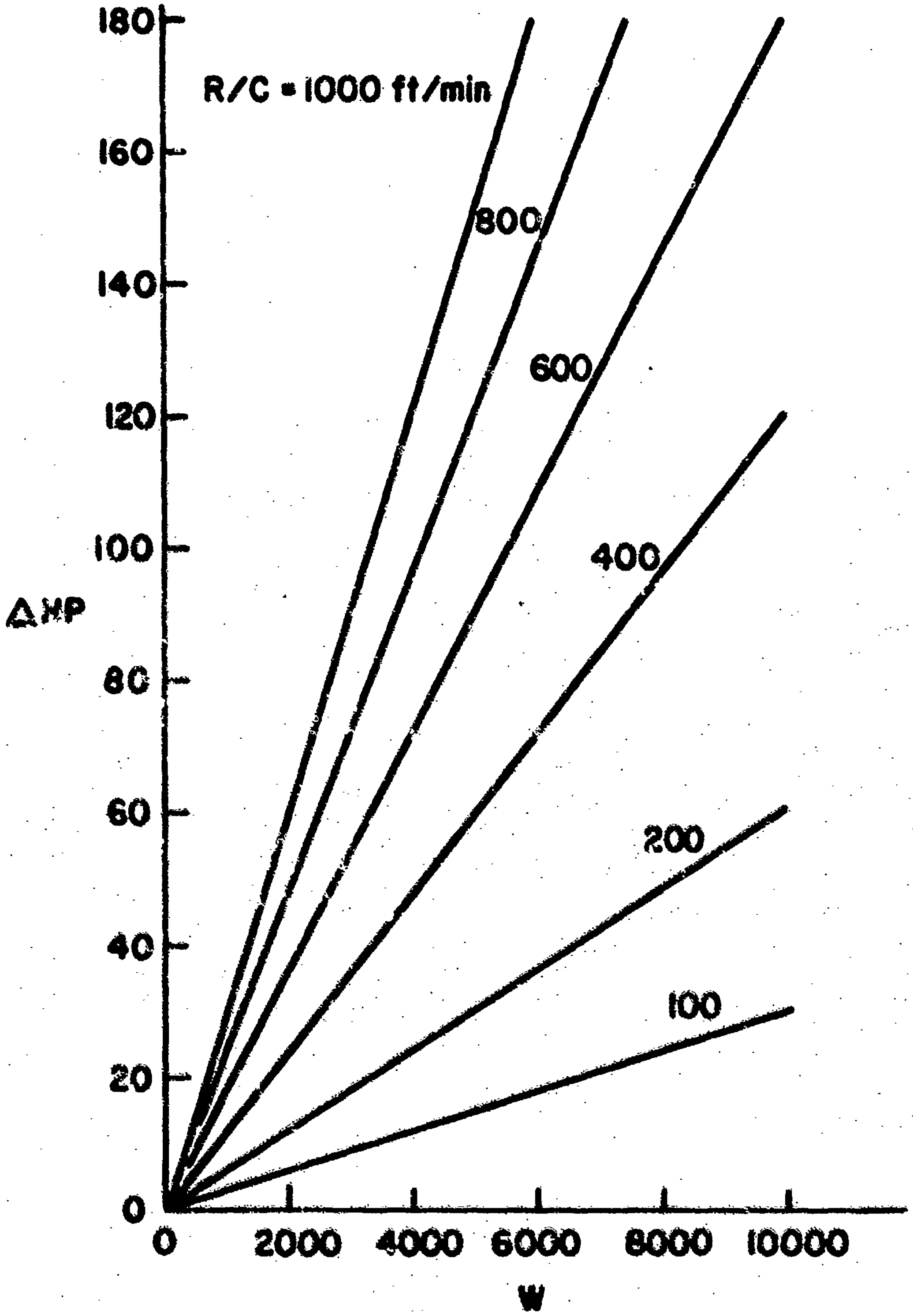

Figure 21. Additional Horsepower vs Weight for Various Rates of Climb $(0 \leq W \leq 10{,}000)$

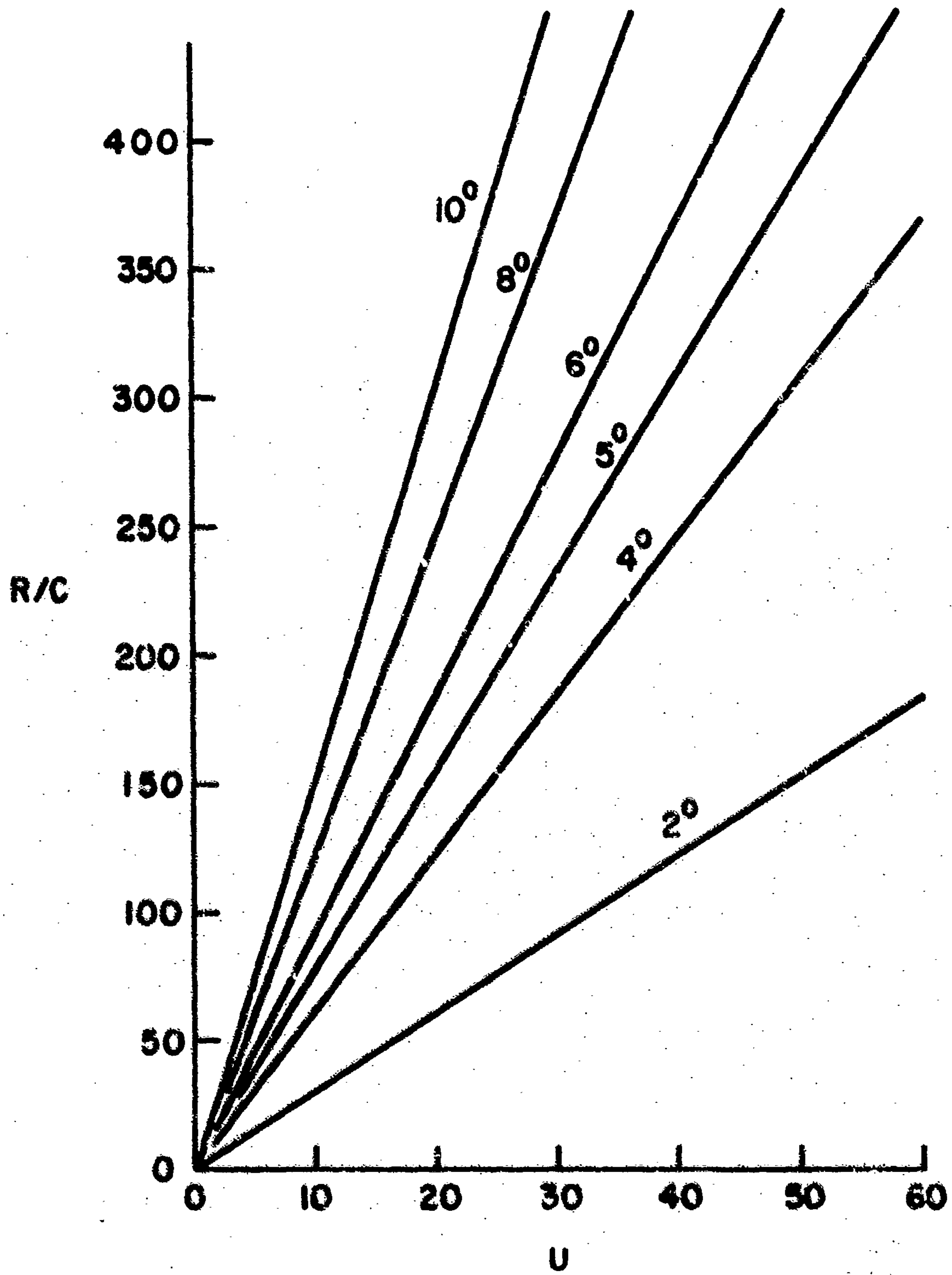

Figure 22. Rate of Climb vs Flight Velocity for Various Angles of Climb

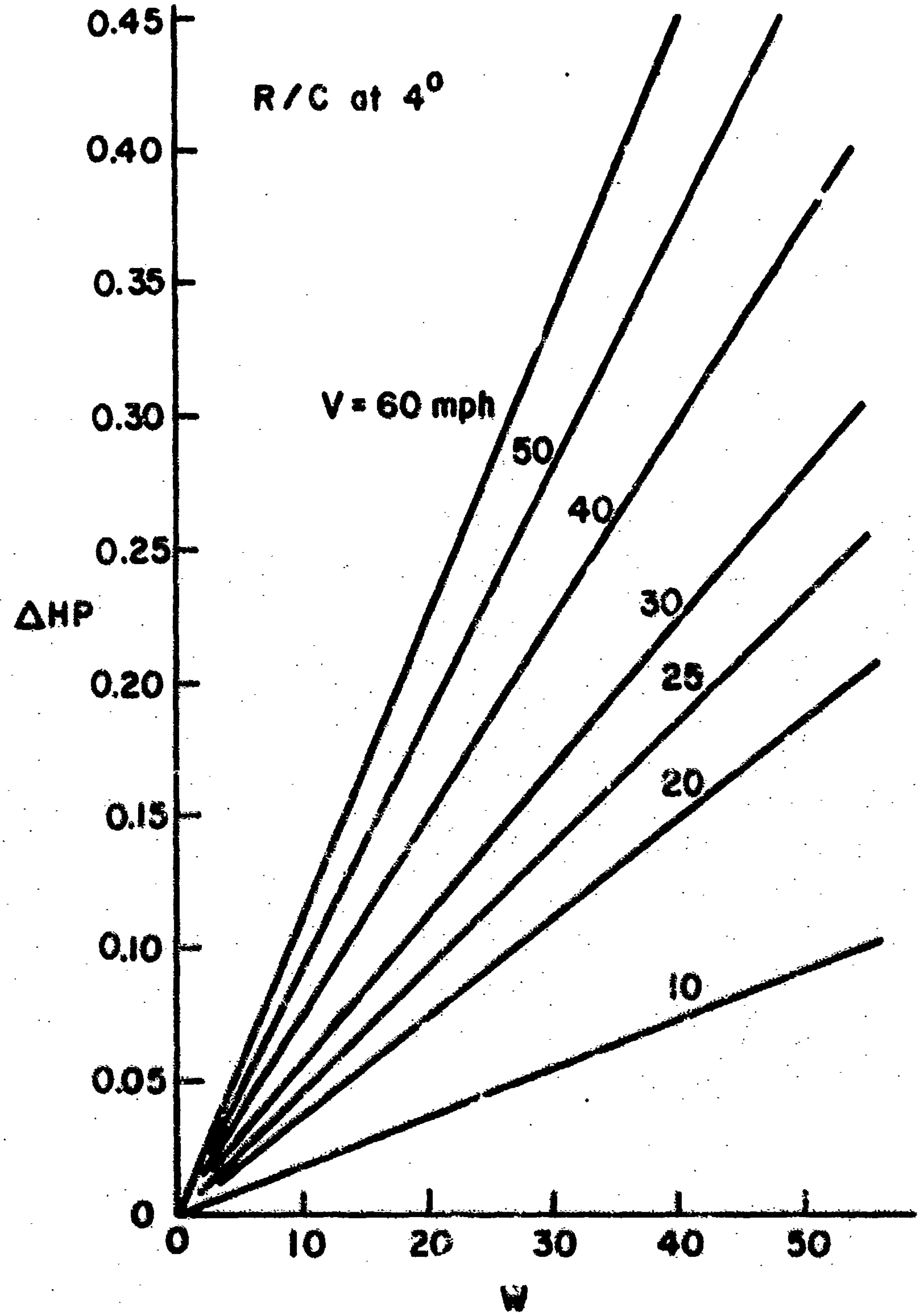

Figure 23a. Additional Horsepower vs Weight for an Angle of Climb of 4° ($0 \leq W \leq 50$)

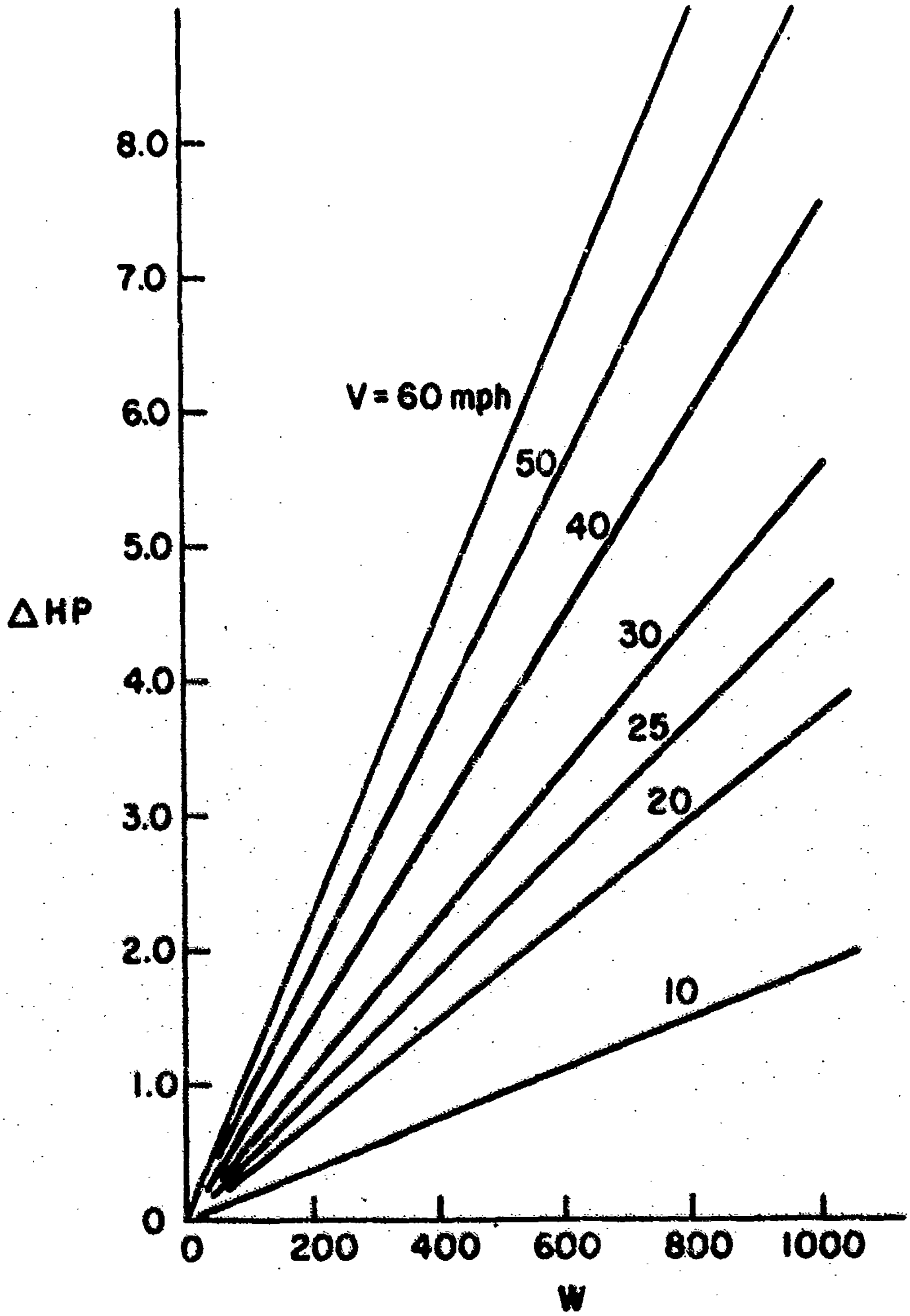

Figure 23b. Additional Horsepower vs Weight for an Angle of Climb of 4° ($0 \leq W \leq 1{,}000$)

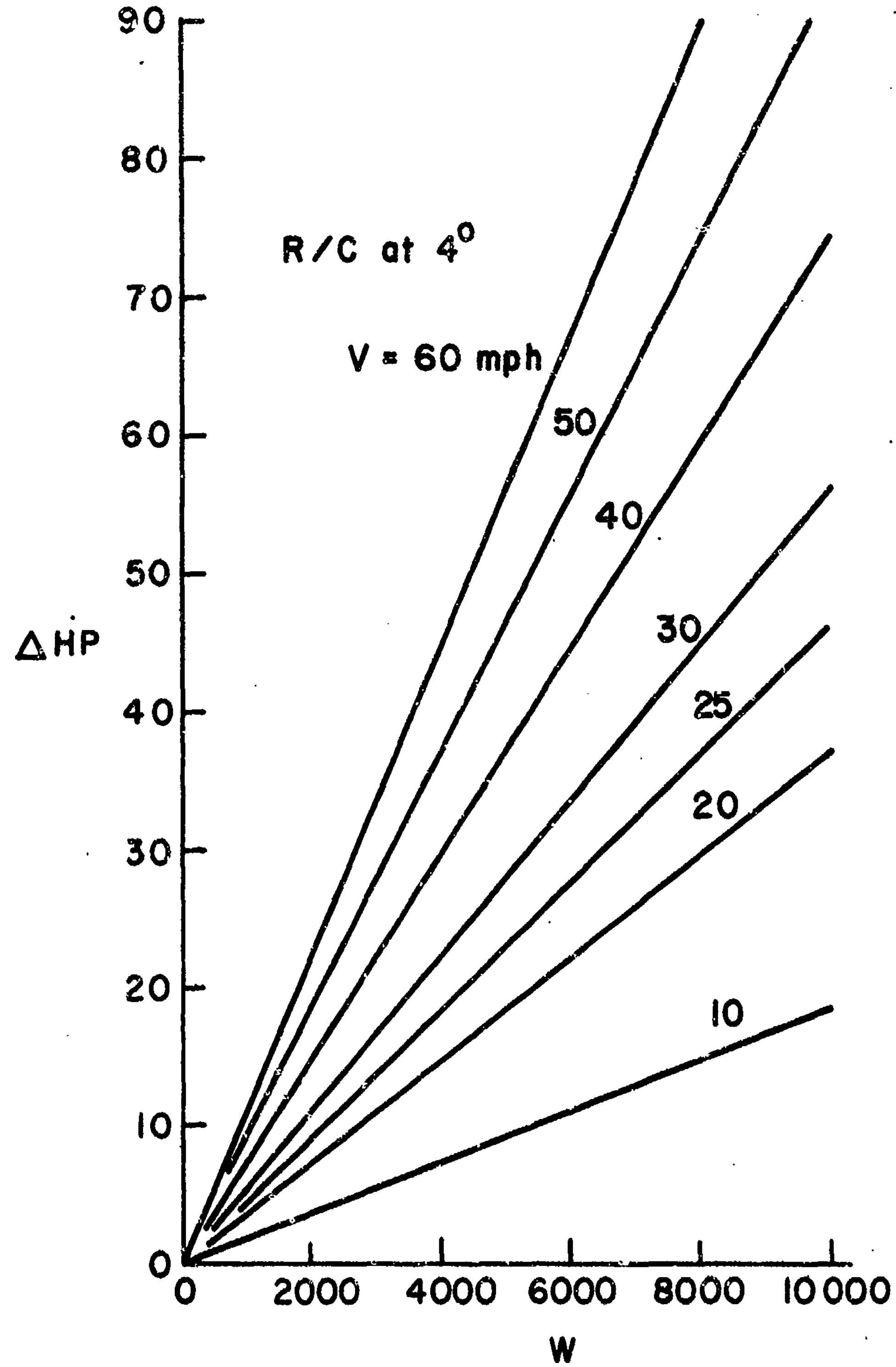

Figure 23c. Additional Horsepower vs Weight for an Angle of Climb of 4^0 ($0 \leq W \leq 10{,}000$)

θ	$R_{@L/D=2.5}$	$R_{@L/D=3}$	$R_{@L/D=3.5}$	$R_{@L/D=4.0}$	$R_{@L/D=4.5}$	$R_{@L/D=5.0}$
0	1	1	1	1	1	1
5	1.0310	1.0252	1.0211	1.0180	1.0155	1.0136
6						
7						
8						
9						
10	1.0542	1.0426	1.0344	1.0282	1.0233	1.0195*
11						
12						
13						
14						
15	1.0694	1.0521	1.0398*	1.0306*	1.0234*	1.0176
16						
17						
18						
19						
20	1.0765*	1.0537*	1.0374	1.0252	1.0157	1.0081
25	1.0753	1.0471	1.0270	1.0129	1.0002	.9916
30	1.0660	1.0326	1.0088	1.9910	.9771	.9660
35	1.0485	1.0104	.9830	.9626	.9466	.9339
40	1.0231	.9802	.9496	.9267	.9088	.8918
45	.9899	.9428	.9091	.8838	.8642	.8485
50	.9491	.8981	.8616	.8343	.8130	.7960

* Approximate best value.

Figure 24a. Ratio of $T_{\theta=0}$ to $T_{\theta\neq 0}$ at Various L/D

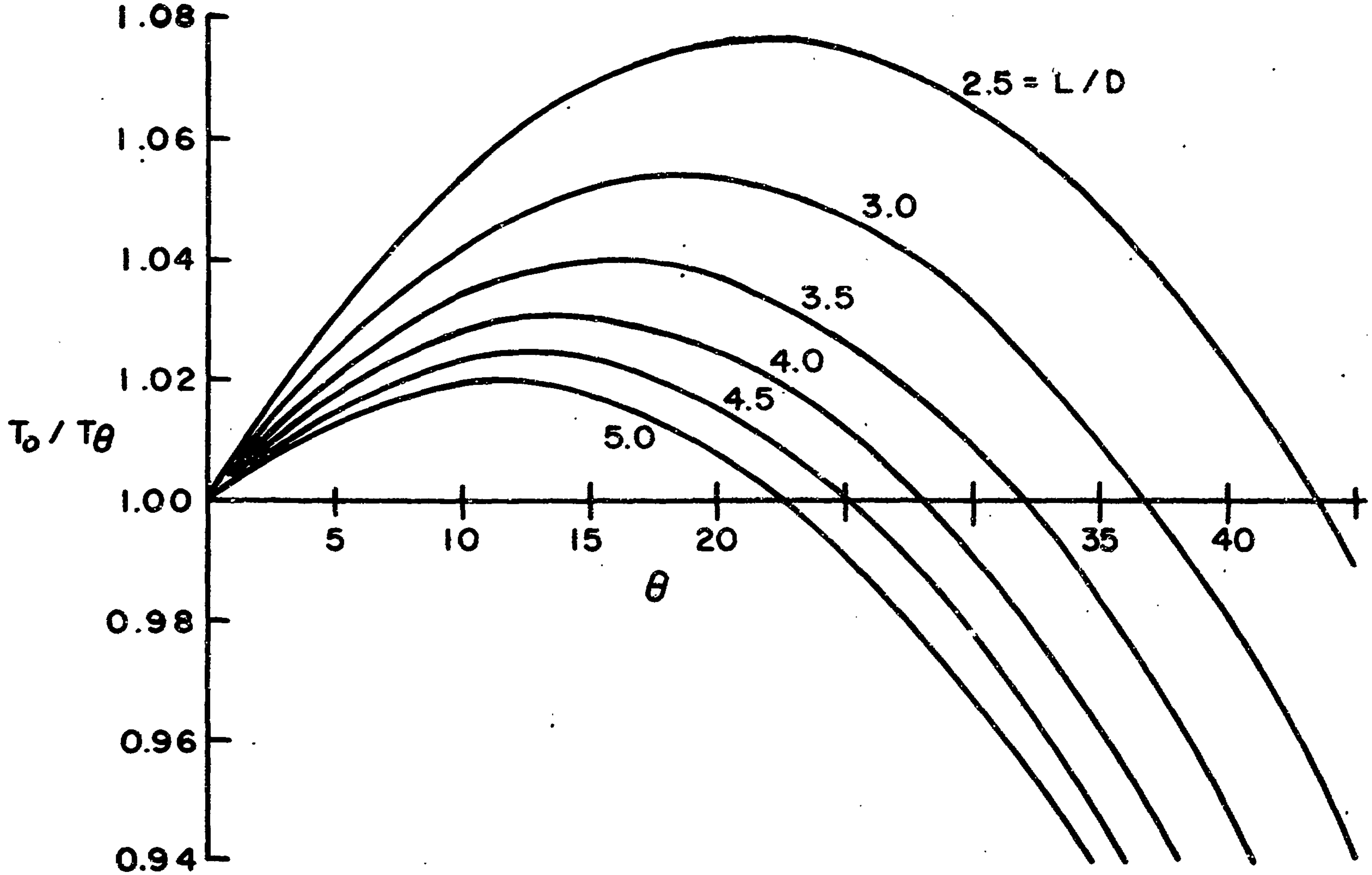

Figure 24b. Thrust Ratio Versus Thrust Angle for Various Lift-to-Drag Ratios

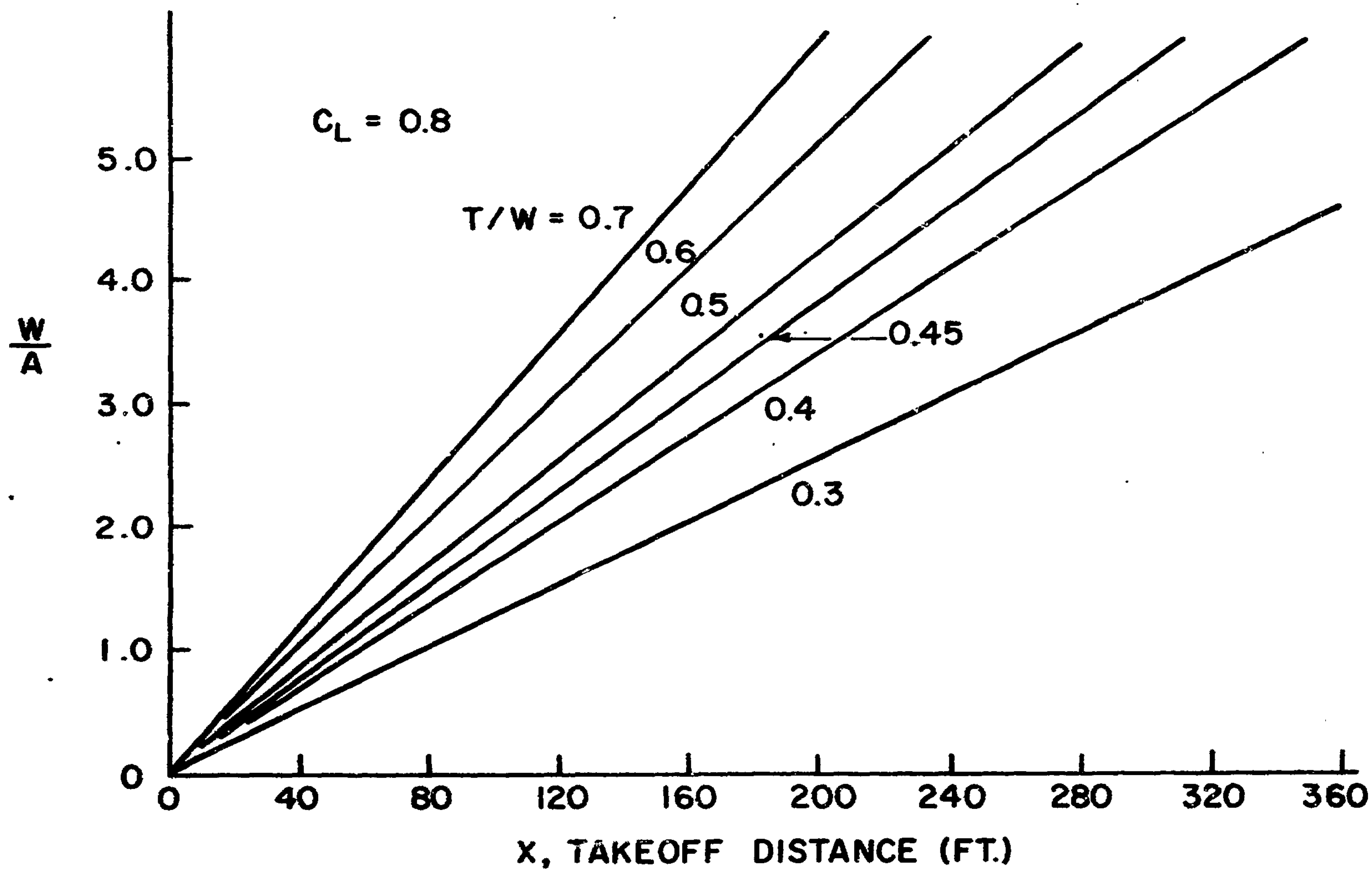

Figure 25. Wing Loading vs Take-Off Distances at Various T/W, (C_L = .8)

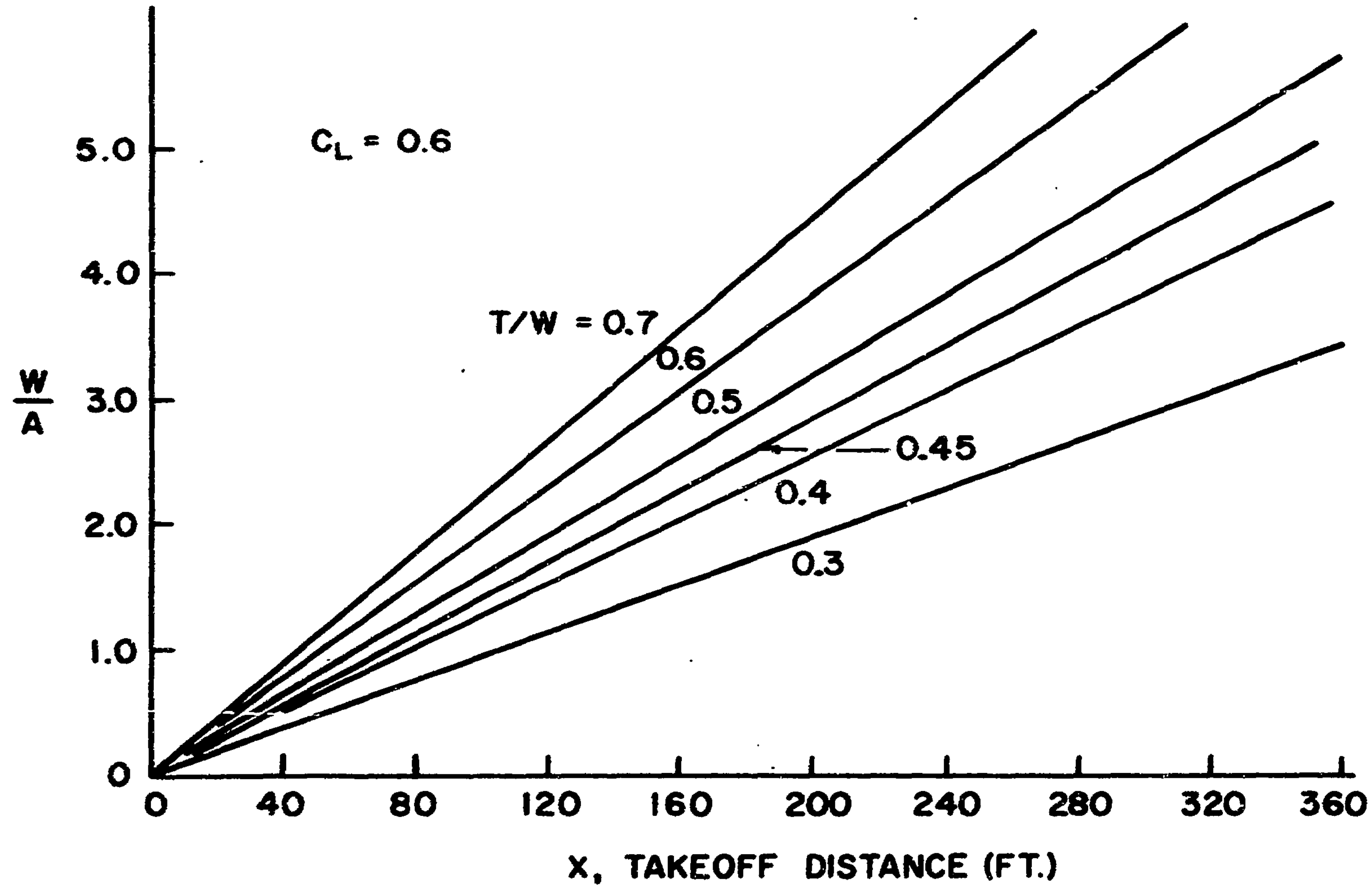

Figure 26. Wing Loading vs Take-Off Distances at Various T/W, (C_L = .6)

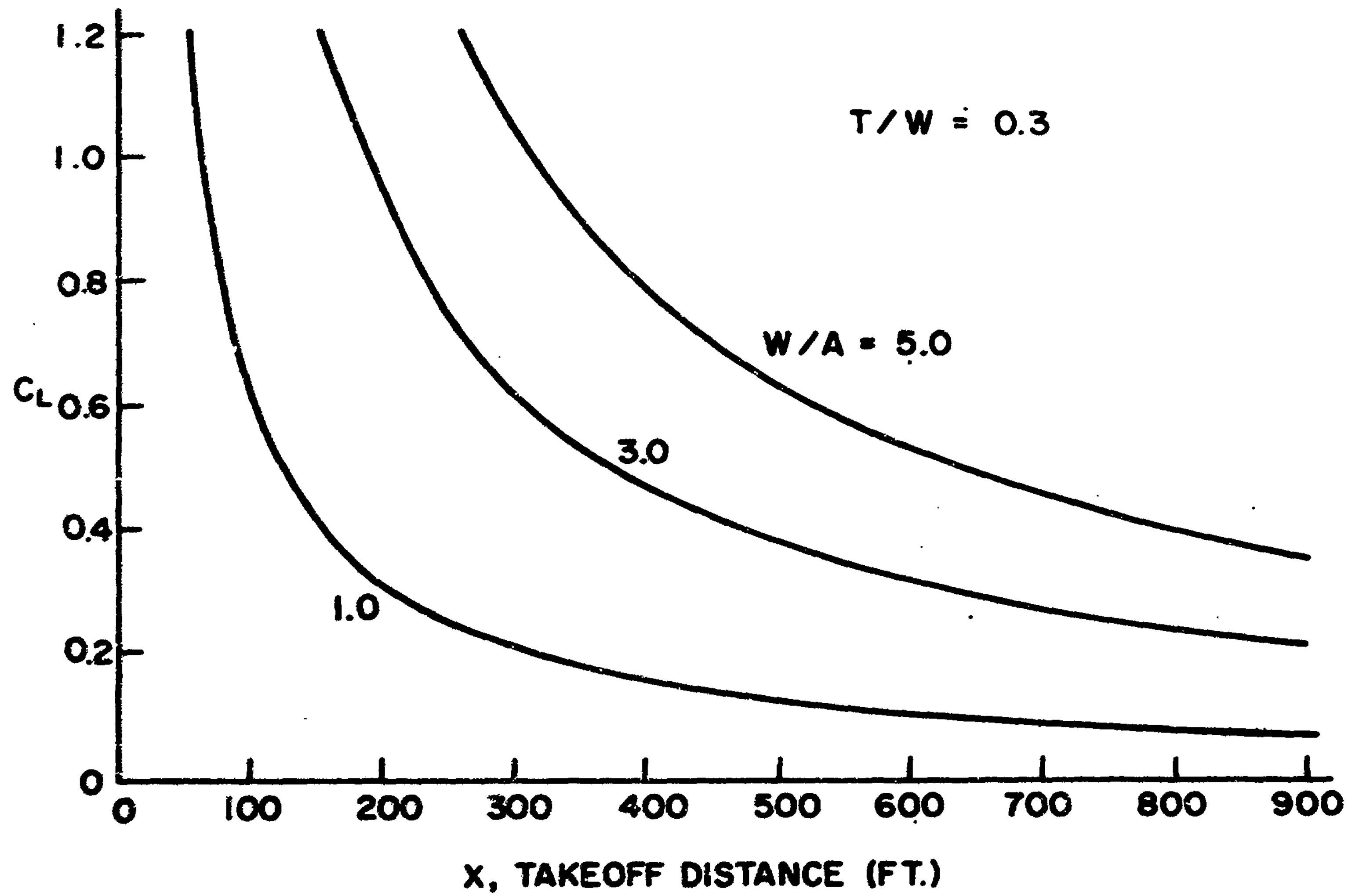

Figure 27. Lift Coefficient vs Take-Off Distances for Various Wing Loadings (T/W = 0.3)

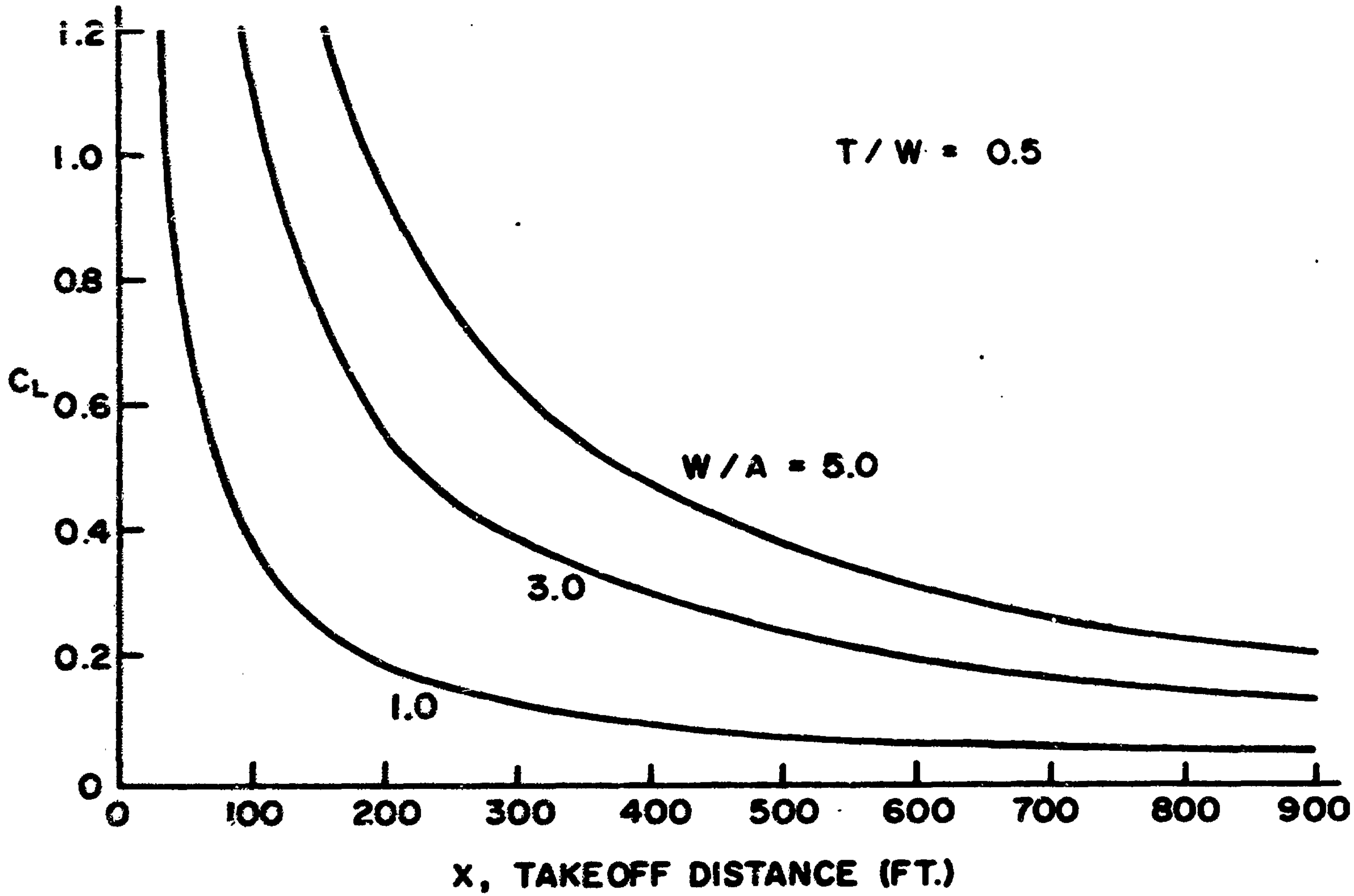

Figure 28. Lift Coefficient vs Take-Off Distances for Various Wing Loadings (T/W = 0.5)

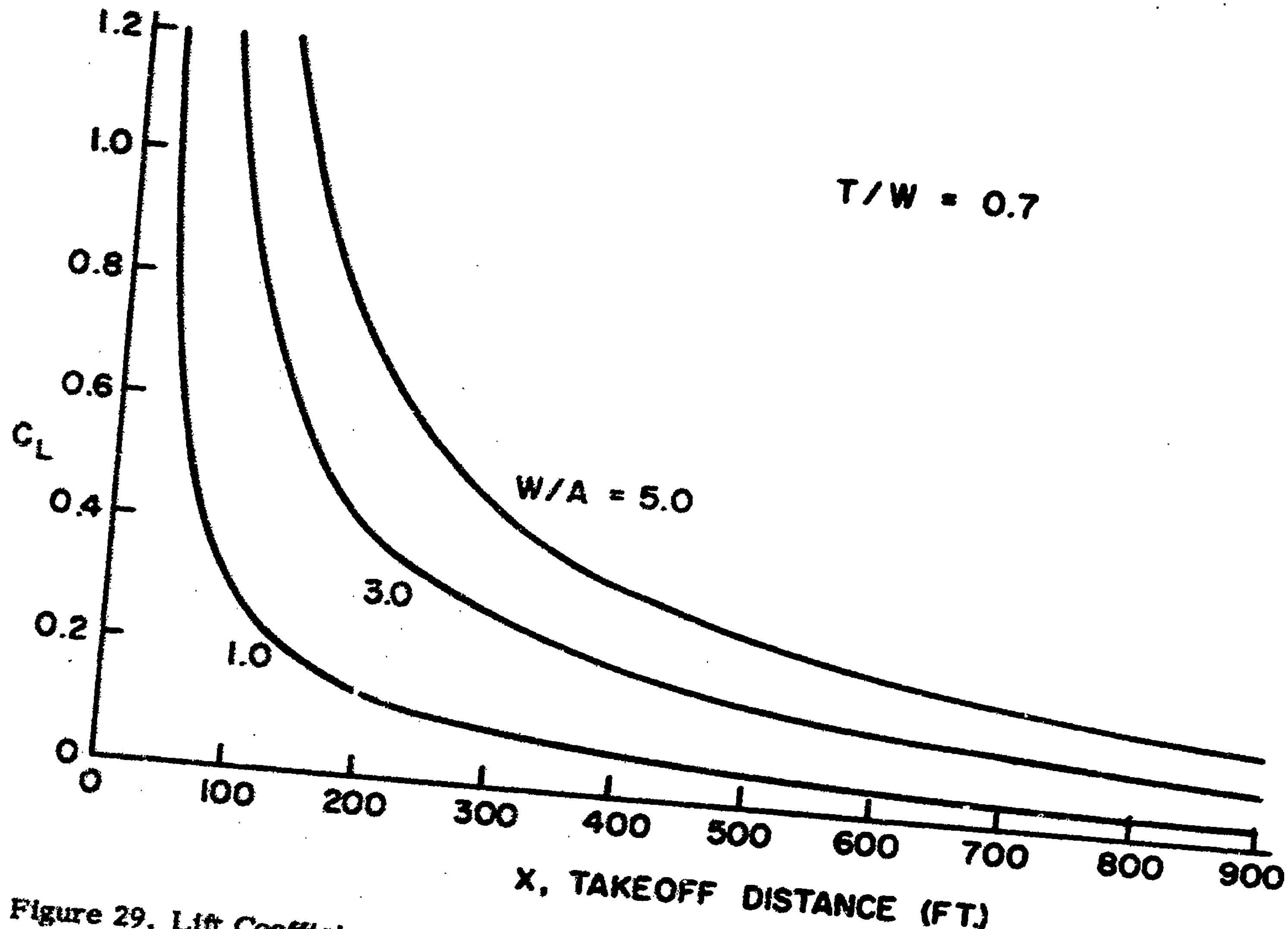

Figure 29. Lift Coefficient vs Take-Off Distances for Various Wing Loadings (T/W = 0.7)

Case	1	2	3	4	5	6	7
D	0	0	≠0	≠0	0	≠0	≠0
R	0	0	0	0	≠0	≠0	≠0
η	1	.8	1	.8	1	.8	.8
L/D	0	0	0	0	3	3	2
μ	0	0	0	0	.1	.1	.1
$\left(\frac{X_T}{X}\right)_{\text{Approx.}}$	1	1.25	2	2.5	1.17	3.57	3.125
$\left(\frac{X_T}{X}\right)_{\text{Exact}}$						3.5014	3.063

Figure 30. Increase in Take-Off Distance Due to Drag, Ground Resistance and Thrust Efficiency

Figure 31. Irish Flyer, F4

Figure 32. Irish Flyer in Flight, F4

Figure 33. Irish Flyer in Flight, (F5)

	AIR												UNDERWATER	
Weight, W	540	540	10,000	450	425	400	425	425	425	400	250	250	540 360	Air Water
Area, A	360	405	1,666	360	360	360	360	360	400	400	400	800	360	60
W/A	1.5	1.5	6	1.25	1.18	1.11	1.18	1.18	1.06	1.0	.625	.312	1.0	6
a_T	14°	4°	4°										4°	4°
C_L	.75	.75	.75	.5	.5	.5	.74	.74	.74	.74	.74	.74	.75	.75
C_D	.340	.148	.148		.200	.2	.296	.247	.185	.185	.185		.148	.148
L/D	2.2	5.1	5.1	2.5	2.5	2.5	2.5	3.0	4.0	4.0	4.0	8.0	5.1	5.1
V Ft/Sec	41.0	41.0	82.2	45.9	44.62	43.28	36.7	36.7	34.7	33.7	26.69	18.86	41.0	2.9
V MPH	27.9	27.9	56.0	31.3	30.42	29.5	25.0	25.0	23.7	23.0	18.2	12.85	27.9	2.0
HP	18.3	7.9	293.0	15.05	13.79	12.59	11.3	9.4	6.71	6.14	3.03	1.07	.146	.37
ΔHP*	6.6 6.4°/400**	6.6 6.4°/400	60.6 2°/200	5.45 400	5.15 400	4.84 400	5.2	5.2	5.15	4.84	.76 100	.76 100	.03 100	-
HP_{Total}	25	14.5	353.6	20.5	18.94	17.43	16.5	14.6	11.8	10.9	3.79	1.83	.176	-

($\rho_w = 1.94$)

*$\Delta HP = \frac{W \; R/C}{550 \times 60}$

** Angle of Climb/Wing Area

Figure 34. Summary of Examples